CR 73238

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SD 67-478-3

A STUDY OF SPACE MISSION DURATION EXTENSION PROBLEMS

VOLUME III

30 October 1967

Prepared by

R. B. Carpenter, Jr.
Project Engineer
Mission Extension Studies

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B. J. Staub
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FOREWORD

This is Volume III of a four-volume report. Volumes I and II, which were published on 15 April 1967 and reflect the results of the first six months effort, reported the result of a manned planetary mission requirements analysis. This Volume reflects the results of the subsystems requirements analysis, with particular emphasis on the factors affecting crew safe return.

The material presented herein was developed under participating company funded effort. The intent was to determine the required subsystem configuration necessary to achieve the mission objectives outlined in Volume I of this report. Particular emphasis was placed on the analysis of potential systems unreliability and the selection of a mission concept which, through use of the availability concept, showed promise of achieving a safe and effective planetary flyby and return.

The study was conducted parallel to the Manned Planetary Flyby Mission Study, NAS8-18025 in order to use the mission and systems design developed therein.

The study was conducted during CFY 1967 by the Systems Engineering Management Division of the Space Division of the North American Rockwell Corporation, under Research Authorization (RA) 02195-15400. Documentation of the study was contracted by the Mission Analysis Division, NASA/OART, Ames Research Center, Moffett Field, California under NAS2-4214.

The work was performed under the direction of Roy B. Carpenter Jr., the Project Engineer/Program Manager, within the Systems Engineering Management Group. Substantial contribution were made to this volume by Mr. Charles Fritz, Peter Fono, J. Bell, and Howard Steverson, the following companies and personnel thereof who provided the data at the respective company expense:

1. A. C. Electronics - D. A. Zeimer, et al
2. Aerojet General - C. Teague, et al
3. Allison Division G. M. - J. C. Schmid
4. Bell Aerosystems - T. P. Glynn
5. Collins Radio - R. Zimmer, et al

6.	Dalmo Victor Company	-	R. L. Straley
7.	Eagle Pitcher Company	-	J. R. Lucas, et al
8.	General Time Corporation	-	Fred Schultz
9.	Hamilton Standard Inc. *		J. J. Holsing
10.	Honeywell, Inc.	-	E. L. Ingalls and C. A. Moosbrugger, et al
11.	Marquardt Corporation	-	J. B. Gibbs
12.	Motorola Inc.	-	R. Crouse
13.	Raytheon Company*	-	H. A. Prindle
14.	Simmonds Prec. Prod.	-	W. E. Nelson and W. E. Dunn

*Provided available data only.

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TABLES

ILLUSTRATIONS

I. BACKGROUND TO SYSTEMS ANALYSIS

1.1 OBJECTIVES OF THIS STUDY PHASE

This phase of the study effort was conducted to determine the ability of contemporary hardware to meet the requirements and constraints imposed on systems by long duration space missions utilizing the Availability Concept as a design tool. Long duration space missions include both the extended earth orbital, lunar, and planetary exploration, and is based on the premise that if a planetary mission can be shown to be feasible and safe, then the near earth missions will be at least as safe, or safer.

The Availability Concept has been developed for application to space missions and is described in Section I of Volume I of this report. In summary, it is a design or mission analysis technique that facilitiates the determination of an optimum man-machine relationship. By proper application, mission effectiveness and safety is maximized through establishment of a safe and reasonable balance between systems and missions performance, reliability, maintainability and operability.

It is a basic intent of this aspect of the study to demonstrate the feasibility and desirability of using contemporary hardware, for these longer space missions, which has been tried and demonstrated to a significant level of reliability. The advantages of this approach are obvious to the reliability engineer who must achieve a significant level of confidence in the reliability of the hardware to be used before the mission starts. The demonstration problem may be seen from Figure 1.1 where the assessed reliability for a program hardware is presented as a function of the program progress. The point to be observed is that the actual reliability of the given component or system, is not known with any assurance until very late in the program — well beyond the permissible time to introduce design changes. Further, with the longer missions and the associated reliability requirements, there is not enough time in a program life span to obtain adequate statistical data if no data is available from previous programs.

The fact that deficiencies exist in component reliability need not be a problem, provided they are known before the mission is initiated. This merely results in the need to add a few spares to the list to compensate for these potential weaknesses. The availability concept permits the mission planner to compensate for potential inadequacies.

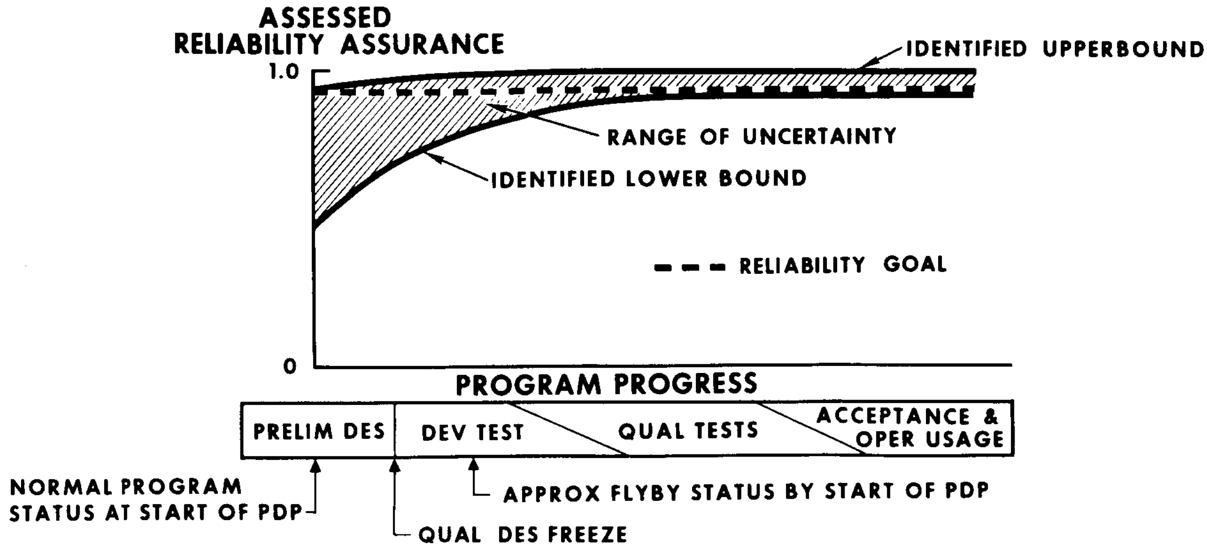

Figure 1.1. Assessed Reliability for Program Hardware as Function of Program Progress

1.2 APPROACH TO THE SUBSYSTEM ANALYSIS

The subsystem requirements and constraints were established through the mission analysis recorded in Volume I of this report. A baseline planetary flyby mission of a duration of 700 days was selected from the alternatives presented under NAS8-18025 (Reference 1.1). Selection was based on the premise that the 1976 – 700-day mission imposed both requirements and constraints equal to, or greater than those required for other Mars and/or Venus missions as well as the extended lunar and earth orbital missions. The earth orbital missions are much less severe in terms of imposed requirements and constraints since both abort and resupply relief are both possible and planned.

The approach to this phase of the study is as reflected in the logic of Figure 1.2. The selected baseline mission provided the subsystem functional requirements and projected duty cycles.

First, a subsystem configuration is evolved, which fulfills the functional requirements, first using Apollo/Saturn hardware, then other contemporary hardware where there was no Apollo equivalent. And finally, where no space-rated hardware exists, new designs were selected with preference being given to those concepts that were closest to the hardware state and showed promise of being reliable.

Next, reliability and crew safety logic diagrams were prepared from the system designs which reflected the critical components in each subsystem function. To accomplish this it was necessary for the designers and reliability engineers to work together and evolve a detailed preliminary design for each system.

The reliability/safety model was then assessed for mission reliability without consideration of maintenance or redundancy, and compared to the mission objectives. Weak links were identified and listed in order of unreliability by system function. Subsequently, they were associated with the prominent failure mode of the involved component and with a specific level of assembly. The resultant was a classification of these problems (weak links) by criticality, probable cause and location within the system.

The Availability Concept developed under Reference 1.2 was then applied to the system and a detailed analysis, as reflected in the logic of Figure 1.3, was performed on each subsystem individually. Reference XXX refers to each block in the system logic and specifically, each weak link, XXX_1 through XXX_n was analyzed in order of unreliability. A fix, or series of fixes, were identified and the results assessed once more for each of the potentially weak links until the probability of crew safe return (P_s) was raised to at least 0.999 for a crew critical system and the risk of failure

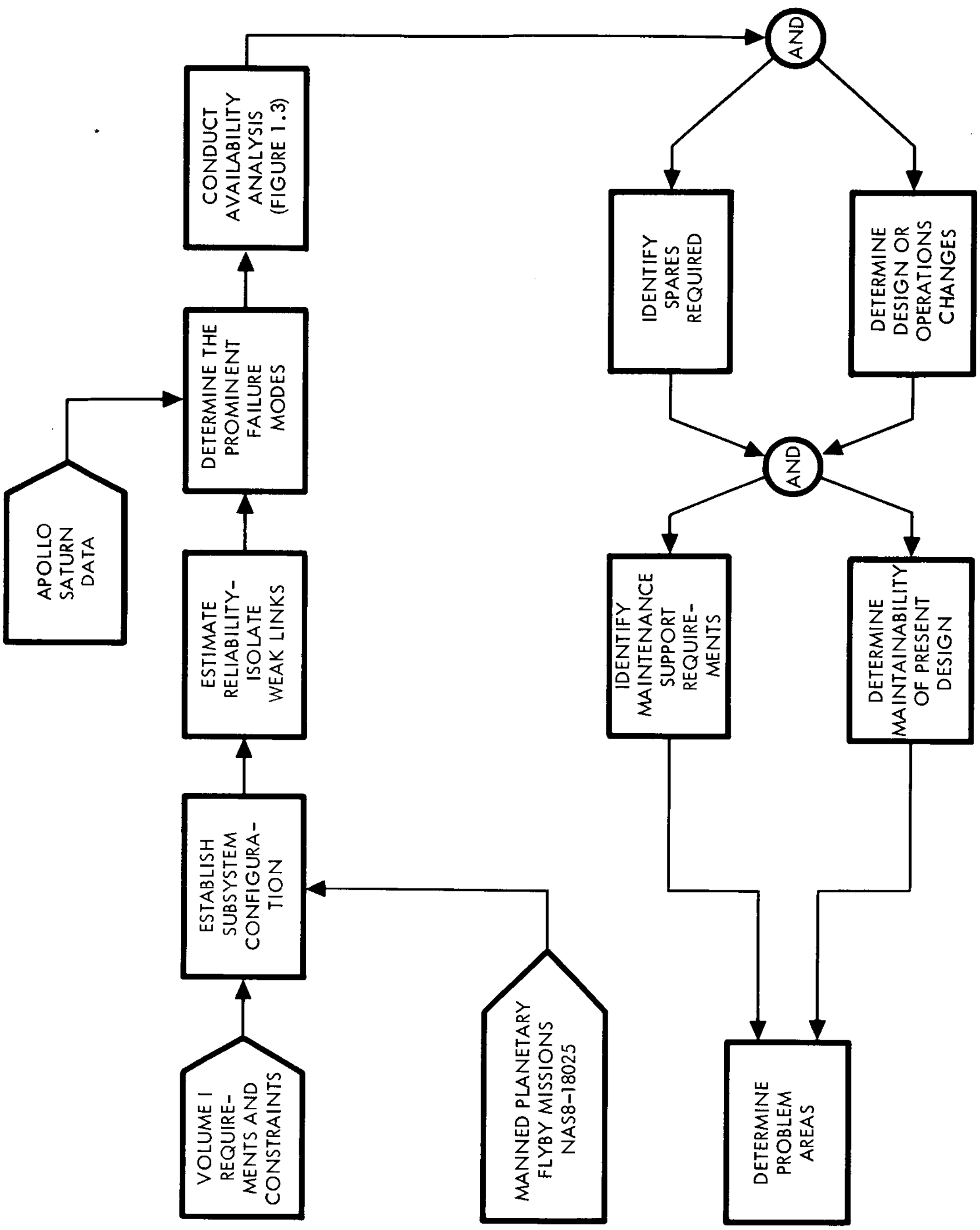

Figure 1.2. Study Approach, Subsystems Analysis

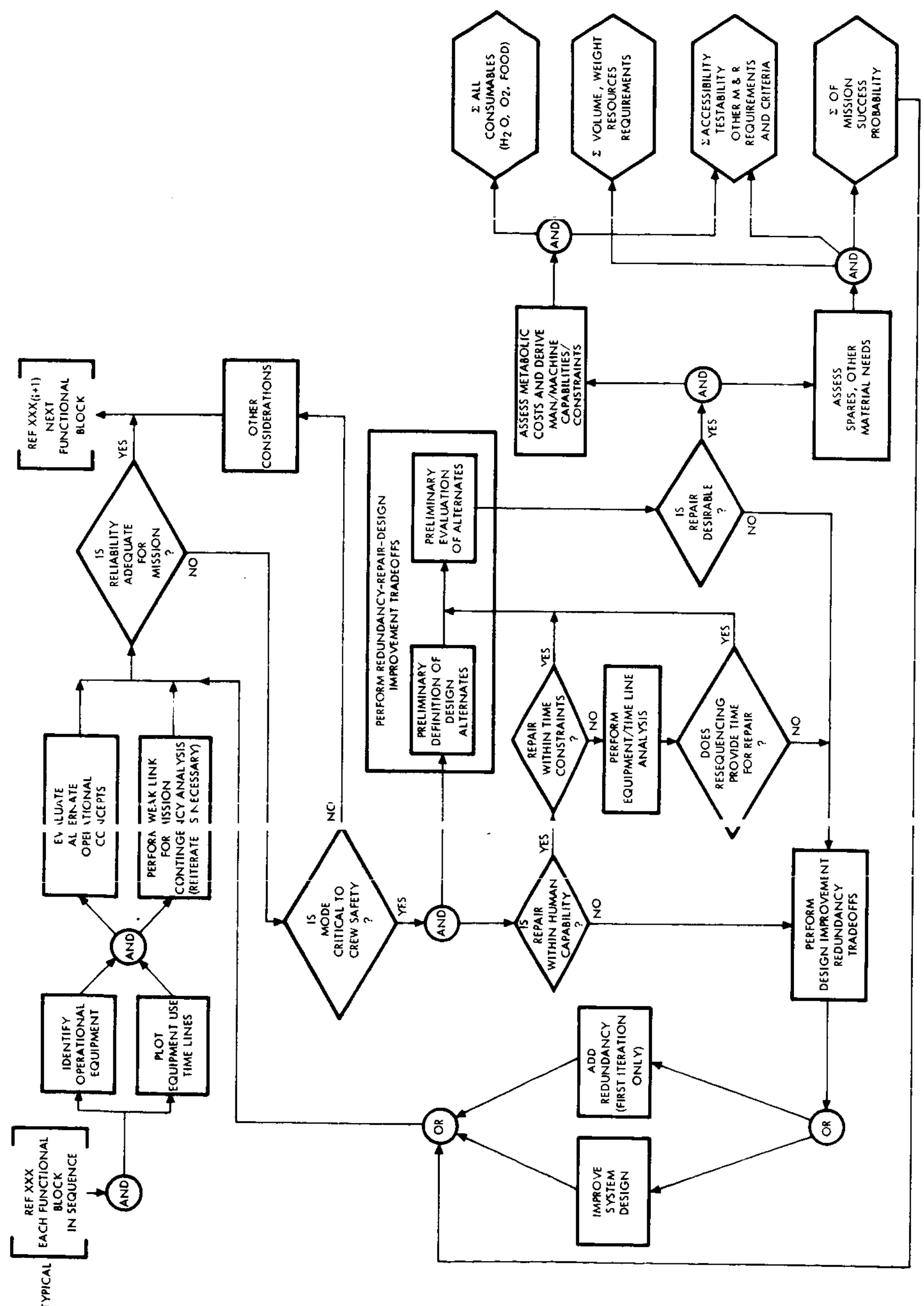

Figure 1.[illegible]. Availability Analysis Logic

(no remaining spares or alternate modes) was approximately equally distributed within the system. For additional details on the technique, see Volume I.

As a result of this analysis, the required spares complement was identified and tabulated by system. Design changes were identified, both for improvement of reliability/safety and to facilitate the required maintenance actions. In addition, the optimum operating mode was identified and associated with the recommended mission/system design.

Subsequently, the design of the present system and components were evaluated to determine the maintainability in its present form, and the redesign requirements to facilitate the required maintenance actions. Many required little or no redesign, or a slight change in the packaging concept.

Finally, each subsystem function was assessed to identify any problem areas which would impose a requirement for further study and/or test programs.

1.3 SUBCONTRACTOR PARTICIPATION

Midway through the study, it became apparent that, in order to perform the proposed analysis to the level of depth necessary for meaningful results, it would be necessary to define the systems design to the point where specific hardware could be identified. To accomplish this task within the scope of the study, it was decided that subcontractors who were known to be expert in the individual fields should be solicited for support in the study. The results were gratifying since the following subcontractors agreed to participate by defining and analyzing to some degree, the system functions listed; each was provided a suggested statement of work along the lines outlined in paragraph 1.2, and performed the study at their expense:

1. A. C. Electronics - Guidance and Navigation
2. Aerojet General - Propulsion Engines
3. AiResearch* - Environment Control and Life Support
4. Allison Div. G. M. - Propellant Tankage
5. Atomics International** - Electrical Power Source

*Taken from several studies, Reference 4.1-1 and 4.1-2, plus consultation.
**Taken from a former funded study, Reference 1.4.

6. Bell Aerosystems - Positive Expulsion Tankage
7. Collins Radio - Communications, Voice and Telemetry
8. Dalmo Victor Corp. - Deep Space and Probe Com. Antenna
9. Eagle-Pitcher - Earth Entry and Peaking Batteries
10. General Time Corp. - Central Timing Equipment
11. Honeywell Corp. - Attitude, Stability and Spin/Despin Control
12. Marquardt Corp. - Reaction Control and Spinup/Despin Engines
13. Motorola Inc. - Up Data Link
14. Raytheon Mfg. Co. - Guidance Computer
15. Simmonds - Propellant Gaging

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II. MISSION OPERATIONS AND CREW SAFE RETURN

2.1 NAVIGATION – EARTH VERSUS ON-BOARD

Apparently there exists two basic modes of executing the navigation function, particularly during the long transplanetary and transearth phases of the mission. During these phases, it seems that it is only necessary to determine the track or trajectory of the spacecraft with respect to that projected for the trip. Where significant deviations are noted, a course or velocity correction is required. These have been estimated to average about 44 meters per second for the Earth-Mars leg and 69 meters per second for the Mars-Earth leg (Reference 1.1). In addition, history from past lunar and planetary programs has shown that these midcourse corrections required were slight, and that the earth tracking capability for these functions was more than adequate.

Assuming that the foregoing is true, transferring responsibility for these functions to the earth complex reduces the operational requirements imposed on such components as the stable platform (IMU) and Guidance Computer (AGC) by thousands of hours. To be specific, for the selected baseline mission, the duty cycle for the IMU (it exhibits the highest failure hazard) drops from 16,800 hours down to a maximum of 825 hours – most of which is during the planetary encounter. This alone results in an increase in reliability from about 0.0973 to about 0.879; or a range of almost certain failure, to only about one chance in ten of failure. In both cases, maintenance and repair was not considered. If repair were contemplated, and the components were impractical to repair in flight, it would require about 6 spares to achieve the 0.99 P_s level for full-time operation, and only one additional for the earth primary mode.

Clearly, since there is no assessible risk in navigation accuracy and a significant improvement in safety, the earth primary mode for the transearth and transplanet phases is the best alternative.

2.2 GRAVITY – ARTIFICIAL VERSUS NONE

Two basic modes of operation are under consideration for the long transplanetary and transearth mission phases which center around the need for artificial gravity. Most of the arguments formulated to date have to do with the physiological aspects of the question. The only consideration given to the operation aspect and its systems engineering aspects are the inference to the "complex functions required to implement artificial gravity." On the contrary, SD studies indicate that the operational and safety factors envolved tend to favor the artificial gravity mode. The results of these analyses are enlarged below.

2.2.1 Improvements in Crew Safety (P_s)

These improvements are realized despite the increased system complexity. See Figure 2.1 where comparison of the logic diagrams for the artificial gravity and zero gravity modes are presented. The artificial gravity mode limits the use of the normal stability control functions and the reaction control engines to the injection, a short zero-g coast phase and planetary flyby phases which amount to a total of approximately 400 hours. The retraction/extension system must operate in various modes for the total time, but much of the system is structural and inherently very reliable. The spin/despin control is a very simple control amplifier, less than 1/100 the complexity of the zero-g stability control system, it also must function for 95 percent of the mission. The engines required to achieve the artificial gravity function continuously for 1.2 hours on 4 different occasions, for a total of 4.8 hours. They are the same as the Marquardt Reaction Control engines and have demonstrated a capability for even longer operation without any failures (Section 4.3).

None of these system functions required to achieve the artificial gravity are complicated, yet their use reduces the duty cycle on more complicated hardware by a factor of over 40. Typical of the uncomplicated nature is the wabble damper which is simply a volume of fluid, the viscosity of which is selected to provide the desired damping action.

For the zero-gravity mode alone, the total stability control functions must operate all of the time, or 16,800 hours. The complexity of the electronics alone assures many failures as demonstrated by the 0.0003 reliability estimate — and this does not take into account the control moment gyro unreliability.

The contrast in effectiveness of the two modes of operation is unavoidable. The artificial gravity mode displays a potential reliability greater than 0.89 without maintenance, where there is virtual certainty of failure in the zero-g mode. Since both modes can be improved through application of the availability concept, those results must also be considered. Through maintenance and repair, both situations can achieve a P_s of over 0.999. However, the zero-g mode requires provisions for over 50 maintenance (M&R)actions and a spare helium supply, where the artificial gravity mode would require only 15 M&R actions.

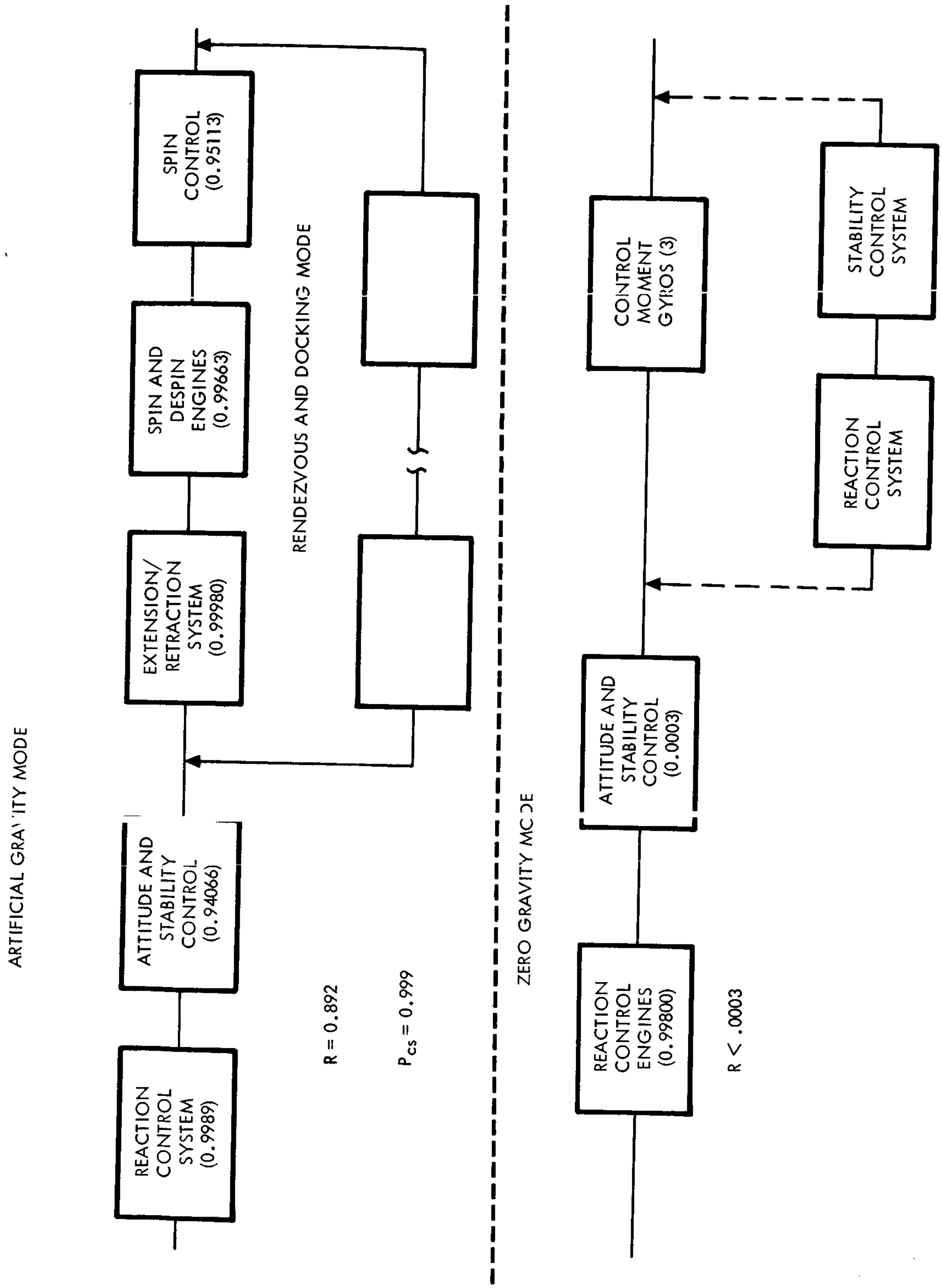


Figure 2.1. Reliability Logic, Stability Control, Artificial vs. Zero Gravity

2.2.2 Other Advantages of Artificial Gravity

These include gravity control of fluids, improved working conditions and efficiency for the crew, a lower electrical power drain and more satisfactory spacecraft/subsystems temperature distribution.

Gravity Control

Fluids in the artificial gravity mode will eliminate the need for use of positive expulsion devices for the major portion of the mission. This, in turn, will substantially reduce the required pressurant and eliminate a major failure mode associated with these systems.

Working Conditions and Efficiency

These factors are improved because man is used to working under a gravitational force. The need for restraints are lessened or eliminated, depending on the forces required. The crew ability to perform useful work increases, as evidenced by the force curves of Volume I section 4.2. The metabolic rates decrease, both in terms of 0_2 consumed and CO_2 rejected, as evidenced by SD studies recorded in Reference 2.1. These studies show that the metabolic rates double under zero-g over 1 "G" for any given task. The effect of an artificial gravity would include a reduction in 0_2 consumption and a decrease in the CO_2 system workload.

Electrical Power

The requirements for power are lower because of the reduced amount of equipments that must be operated. These fall into two categories: those associated with the attitude and stability control functions, which includes part of the guidance functions; and those associated with the need to maintain a thermal balance in some areas. In the latter category it will be necessary to use electrical heaters to maintain fuel temperatures to keep them from freezing in areas not exposed to solar heat (aft sections). In addition, areas of the spacecraft proper may require heating to maintain the desired thermal balance and prevent undesirable stresses. A slow roll only cures part of the problem, even as orientation parallel to the sun's rays only cures part of the problem.

2.3 PROFILE COMPLEXITY INFERENCES

The selected mission profile and its complexity can exercise a pronounced influence on the mission success and safety; the more complex the required series of operations, the less safe the mission will be or the more opportunities for failure or error. This may be seen from a contrast in the reliability expressions for the lunar or planetary flyby mission and the landers. Some of the major operations are given in Figure 2.2; however, an analysis of the associated profiles reveals that there are about 38 discrete operations for the lander missions and only about 16 for the flyby missions.

A discrete operation is a point in the mission profile where status quo must be changed; for example, make a velocity correction or conduct docking operations. In each of these cases the number of spacecraft functions required to accomplish the operation increases by factors of 25 or more. Where, for a coast period, basically, the ECLSS functions affect crew safety, yet, during a simple velocity change, many more system functions enter into the picture. For more stringent activities, such as rendezvous or landing operations, the required functions which influence crew safety proliferate to the point where, for a one-hour operation, the hazard rate may be equal that of the long coast periods.

Consider as an example the following:

Where N is the number of discrete operations required

$$\text{Mission Reliability } (R) = e^{-(\lambda_1 t_1 + \lambda_2 t_2 + \ldots + {}_n t_n}$$

If the failure hazard of the coast phase λ_c was taken as the reference baseline then the Reliability (R_c) is:

$$R_c = e^{-\lambda_c t_c}$$

and using the simplifying assumption that 25 λ_c represents the approximate failure hazard for the average discrete operation which is known to be conservative, then the failure hazard for a 4-hour rendezvous operation would be: 100 λ_c. The increase in failure hazard for the lander mission would be: $(100\lambda_c)$ (38 - 16) = $2200\lambda_c$. Since the total failure hazard for the transearth phase of the baseline mission is only $3300\lambda_c$, it is evident that the mission profile complexity should be held to a minimum to maximize crew safe return (P_s).

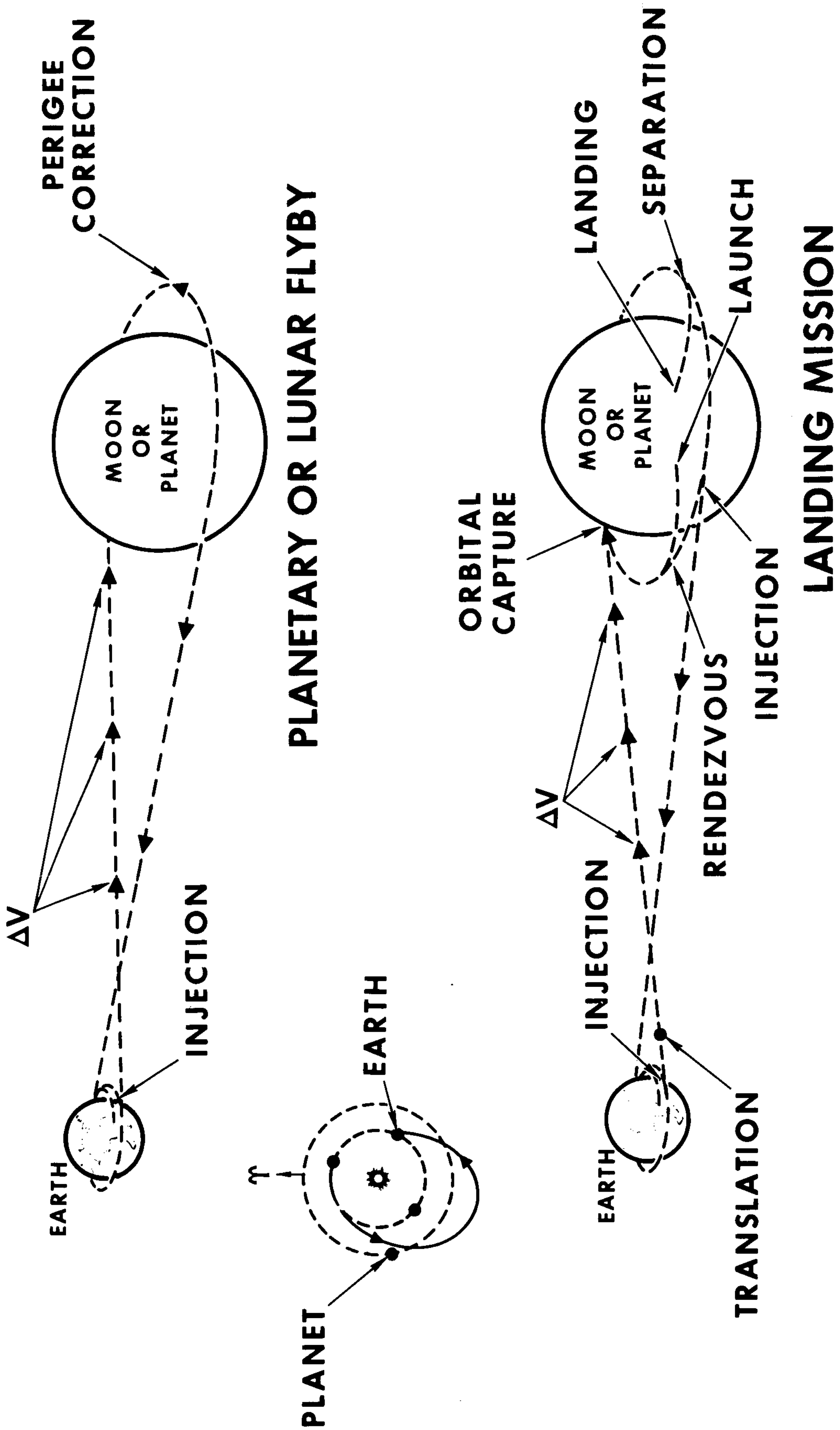


Figure 2.2. Contrast of Mission Complexity, Flyby vs. Landers

Table 2.1 shows in a relative sense the results of a comparison of the various potential planetary mission modes and their respective effects on both mission success and crew safe return. A third column has been added to reflect the potential effects and relative hazard associated with partial crew losses. The baseline planetary flyby mission was used as the reference for the relative estimates.

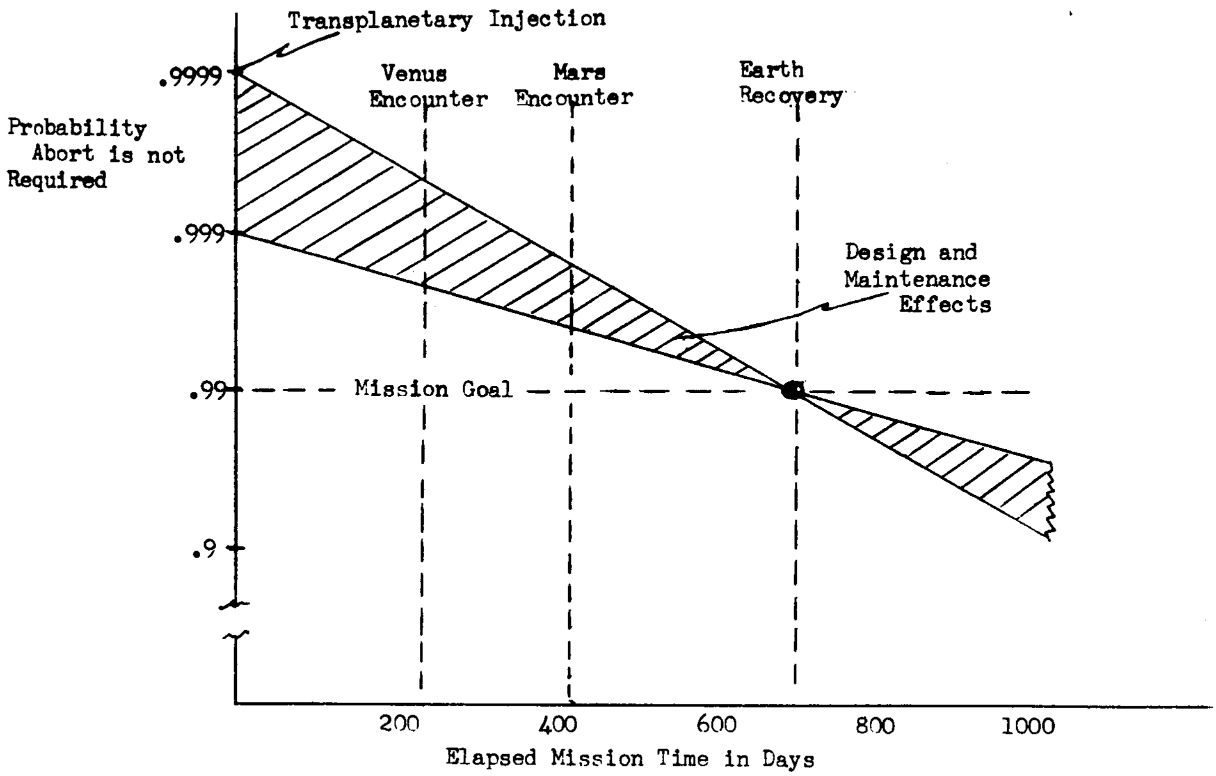
Table 2.1. Relative Success/Safety of Potential Planetary Missions

Mission Class	Relative Failure Hazard	Relative Safety Hazard	Relative Hazard of Partial Crew Loss
1. Planetary flyby	1	1	1
2. Flyby with MSSR	80	20	60
3. Flyby with aerobraker ELF	115	90	300
4. Flyby with aerobraker EOF	60	50	135
5. Flyby with retrobraker ELF	100	80	280
6. Flyby with retrobraker EOF	50	40	120
7. Aerobraker with MEM	110	90	440
8. Aerobraker vehicle	30	25	200
9. Retrobraker vehicle with MEM	105	80	405
10. Retrobraker vehicle	25	20	160

2.4 ABORT CAPABILITY AND CREW SAFETY

Use of injected weight in an optimum manner is a key factor in assuring the safe return of a planetary exploration party. This is particularly true since there is a very limited capability for missions through the 1970's. It has been shown in former studies and others (References 1.1 and 1.2) that the probability of safe return (P_S) for these missions is proportional to the weight devoted to improving the system effectiveness through maintenance. As an example, the Boeing study (Reference 2.2) shows that P_S extends from 0.90 to 0.99 with about 1800 kg or 818 pounds of spares; the SD study (Reference 1.2) confirmed this, (Figure 2.4).

Probability of Abort as a Function of Mission Phase



Improving Crew Safety Through Reallocating Propellant Weight

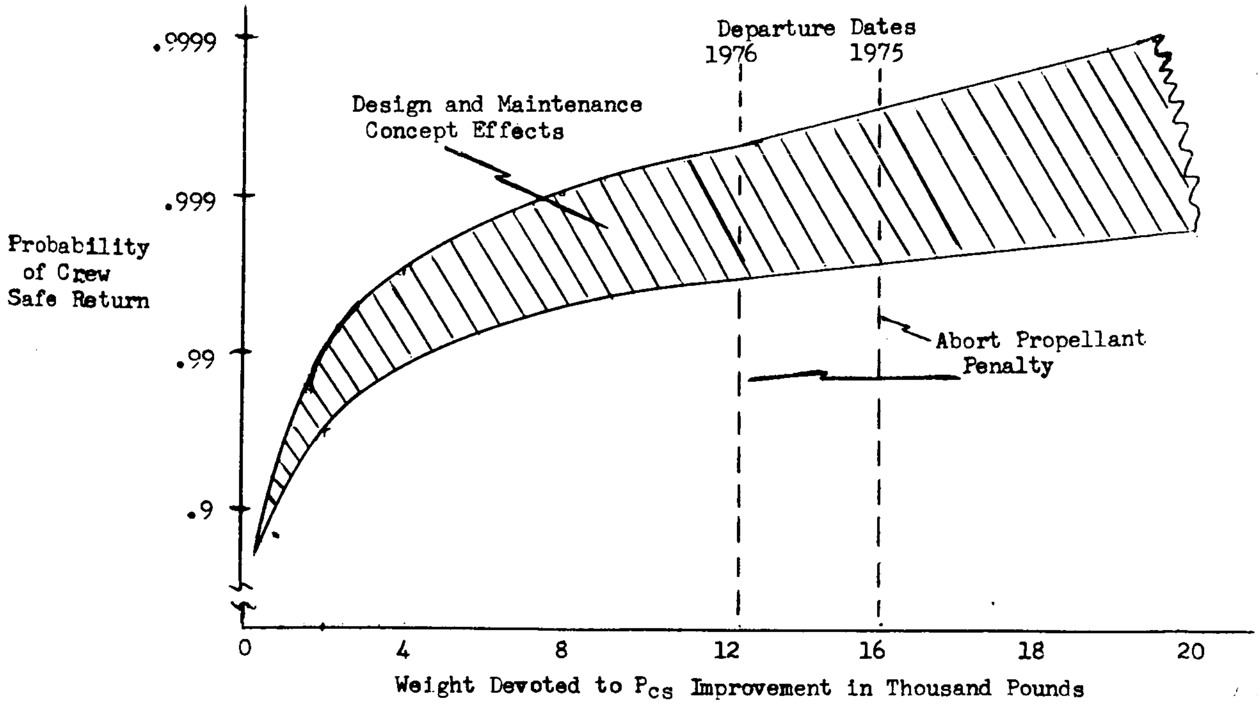


Figure 2.3. Considerations Toward Crew Safety, Abort Vs. Maintenance

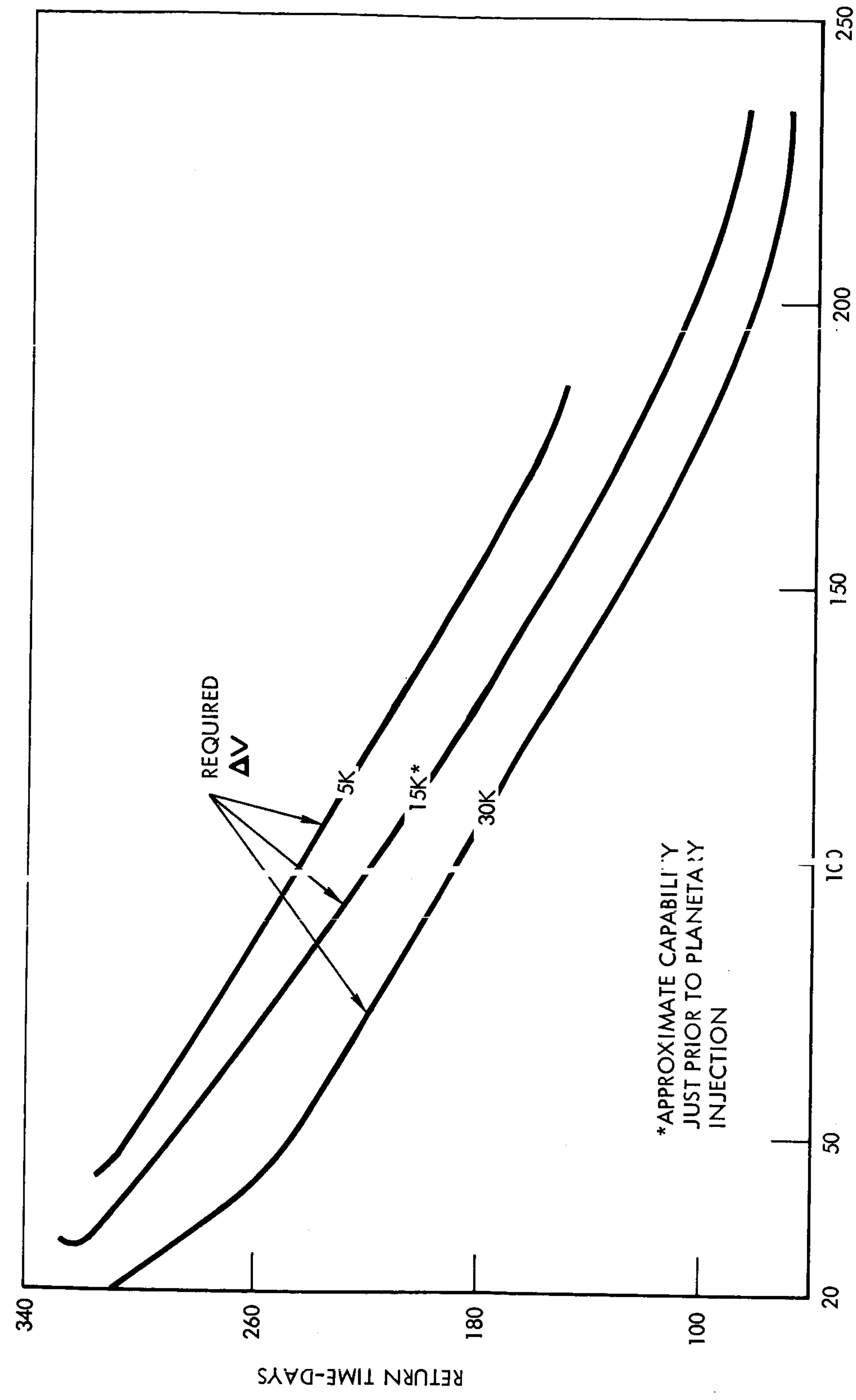


Figure 2.4. Velocity Change and Transit Time Requirements for Abort from Planetary Mission

Since optimum use of weight infers use where the most return in terms of gain in P_s is possible, it seems evident that weight devoted to crew safety functions should be carefully allocated. Since abort is provided to augment crew safety, any capability devoted to this function should be evaluated against other potential uses, as well as the requirement. Figure 2. 3 presents the projected probability that abort is not required as a function of elapsed time into the 1976 baseline mission. Note that if a spacecraft is injected into the transplanetary phase on a mission designed for a 0. 99 probability of safe return, the probability of having to abort is lowest just after injection. Prior to injection abort from earth orbit is possible without penalty. In contrast to the requirement, the abort capability is only useful for the first few minutes. The results of NAS8-18025 suggests that only about 3 minutes were practical; after that, the ΔV required and the associated transit time were prohibitive. In a former SD study (NAS9-3499 reference 1. 2) as shown by Figure 2.4 , the transit time was shown to be high (hundreds of days) even if there was sufficient propellant on board. This all adds up to the fact that the safest way home is via the planned trajectory.

Given that abort is a somewhat dubious picture, the question of most effective use of that weight must be resolved. From Figure 2. 4, it can be seen that using the weight allocated for a three-minute abort capability by NAS8-18025 for improving crew safety through maintenance and repair (M&R) can raise the probability of safe return (P_s) for the crew by one to two, or more, orders of magnitude.

The results of this analysis indicates that more effective use of any weight surplus can be used for improving P_s through M&R and design precautions rather than through provisions for post injection abort.

III. MAINTENANCE REQUIREMENTS ESTIMATION CONCEPT

3.1 THE SPARES ALLOCATION PROCESS

It has been reasonably well established that maintainability will be a key factor in the use of contemporary systems for long duration space missions. Any realistic maintenance plan depends upon the provision of standby spares for the replacement or repair of failed equipment items. For a maintenance plan to be effective, the number and levels of spares provided must be logically developed from characteristics of the system and subsystems involved, and requirements of the mission which they are to perform. The "Availability Concept," as described and discussed in Volume 1.0 of this report and reference 1.2, is a procedure or methodology specifically developed to determine subsystem maintainability requirements for long duration space missions.

This section will concern itself primarily with the derivation of maintainability requirements and the development of a preliminary mission maintenance plan designed to improve the probabilities of safe return (P_s) and mission success to acceptable levels on the basis of a balanced or equalized risk throughout a given criticality class.

Assuming that reliability logic diagrams, equipment failure rates, equipment duty cycles, etc., are available as results of the first steps in the availability analysis, the accomplishment of the maintenance requirements analysis depends upon the ability to evaluate the effects of providing different numbers and levels of spares for the various potentially weak equipments. The system configuration which is least difficult to evaluate is the one in which the total reliability logic diagram consists of single equipment items in series as indicated in Figure 3-1. For purposes of illustration let there be ten equipment items, each with a failure rate, λi, of 1.0 failures/10^6 hours of operation; and a duty cycle, t_i, of 10^4 hours/mission. The total system reliability for such a system-mission combination may be determined by use of the general Equation 3.1 which is P_s.

$$P_s = \prod_{i=1}^{N} (R_i) = R_A \times R_B \times R_C \times \ldots \times R_N \qquad (3.1)$$

It is generally accepted that if an equipment item is operating within its normal operating life, i.e., equipment failures will be random in nature and not due to deterioration of components, then the mean-time-between-failures, $MTBF_i$, and thus the random failure rate, $\lambda_i = 1/MTBF_i$, will remain constant and the probability of occurrence, P_x, of x number of failures may be determined from Equation 3.2.

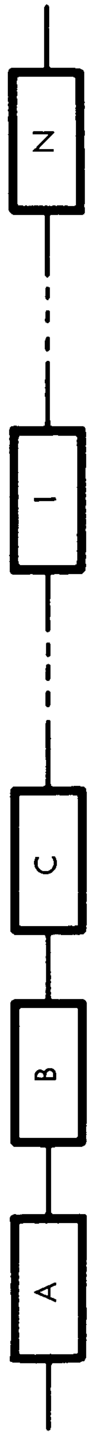


Figure 3.1. Single Equipment Items, in Series

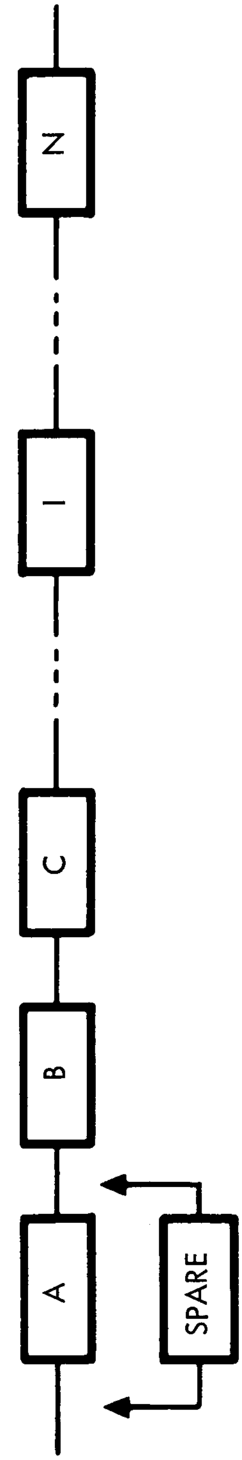


Figure 3.2. Effect of Providing Single Spare Unit for Item A

$$P_x = \sum_{j=1}^{x} \frac{(\lambda_i \; t_i)^j}{j!} (e^{-\lambda_i t_i}) \qquad (3.2)$$

If no sparing capability has been provided for any of the items, x, in each case becomes zero and the reliabilities become

$$R_i = e^{-\lambda_i t_i} \qquad (3.3)$$

The mission system P_s for this first illustrative example as computed by use of Equations 3.1 and 3.3 with the values given is:

$$P = \prod_{i=1}^{N} (R_i) = \prod_{i=1}^{10} (e^{-\lambda_i t_i})$$

$$= (e^{-0.01})^{10} = (0.9900)^{10}$$

$$= 0.9045$$

The effect of providing a single spare unit for Item A, as indicated in Figure 3.2, would be that of replacing Item A with some equivalent, Item A1, with the same failure rate and duty cycle, but which had the characteristic of not affecting the performance of the total system until it experienced a second failure. For such an equivalent item, the x of Equation 3.2 becomes 1, and the reliability for Item A1 may be determined from Equation 3.4.

$$P_{s1} = e^{-\lambda_{A1} t_{A1}} + \lambda_{A1} t_{A1} (e^{-\lambda_{A1} t_{A1}}) \qquad (3.4)$$

Using both Equations 3.3 and 3.4 with Equation 3.1 and the values given, the mission system P_{s2} becomes:

$$P_{s2} = R_{A1} \times \prod_{i=2}^{N} (R_i)$$

$$= \left[e^{-\lambda_{A1} t_{A1}} + \lambda_{A1} t_{A1} \left(e^{-\lambda_{A1} t_{A1}}\right) \right] \left(e^{-\lambda_i t_i}\right)^9$$

$$= [0.9900 + 0.00995] (0.9900)^9$$

$$= [0.99995] (0.9136)$$

$$= 0.91355$$

From this it can be seen that adding the capability to nullify the first failure of Item A would increase the reliability of Item A from 0.9900 to 0.99995 and the reliability of the total system from 0.9045 to 0.91355.

In a similar manner, providing the same single spare unit to Item B would increase the mission system P_{s3} to:

$$P_{s3} = \left[e^{-\lambda_{A1} t_{A1}} + \lambda_{A1} t_{A1} \left(e^{-\lambda_{A1} t_{A1}}\right)\right]$$

$$\times \left[e^{-\lambda_{B1} t_{B1}} + \lambda_{B1} t_{B1} \left(e^{-\lambda_{B1} t_{B1}}\right)\right] \times \prod_{i=3}^{10} \left(e^{-\lambda_i t_i}\right)$$

$$= \left[0.99995\right]\left[0.99995\right](0.9900)^8$$

$$= 0.999278$$

The ultimate effect of providing "first failure nullification" capability to this system would obviously be providing it for all ten items. In such a case, the P_s would be raised to:

$$P_4 = \prod_{i=1}^{10} \left[e^{-\lambda_{i1} t_{i1}} + \lambda_{i1} t_{i1} \left(e^{-\lambda_{i1} t_{i1}}\right)\right]$$

$$= \left[0.9900 + 0.00995\right]^{10}$$

$$= \left[0.99995\right]^{10}$$

$$= 0.99950$$

It is is desirable to improve the ultimate P_s to some higher level, a second standby spare or "second failure nullification" capability for some or all of the other items may be added. Providing such a capability to the first item, for example, and extending Equation 3.2 to Equation 3.5

$$R_{A2} = e^{-\lambda_{A2} t_{A2}} + \lambda_{A2} t_{A2} \left(e^{-\lambda_{A2} t_{A2}}\right) + \frac{(\lambda_{A2} t_{A2})^2}{2} \left(e^{-\lambda_{A2} t_{A2}}\right) \quad (3.5)$$

the mission system P_s is increased to:

$$P_{s5} = R_{A2} \times \prod_{i=2}^{10} \left[e^{-\lambda_{i1} t_{i1}} + \lambda_{i1} t_{i1} \left(e^{-\lambda_{i1} t_{i1}} \right) \right]$$

$$= [0.9900 + 0.00995 + 0.0000499] (0.99995)^9$$

$$= [0.9999999] (0.99995)^9$$

$$= 0.99955$$

Continuing as before, if the "second failure nullification" capability is added to all ten items, the mission system P_s is improved to:

$$P_{s6} = \prod_{i=1}^{N} \left[e^{-\lambda_{i2} t_{i2}} + \lambda_{i2} t_{i2} \left(e^{-\lambda_{i2} t_{i2}} \right) + \frac{(\lambda_{i2} t_{i2})^2}{2} \left(e^{-\lambda_{i2} t_{i2}} \right) \right]$$

$$= [0.9900 + 0.00995 + 0.0000499]^{10}$$

$$= [0.9999999]^{10}$$

$$= 0.999999$$

In practice it may not be possible or desirable to provide the same number of spare units to all items of a system. The same general Equation 3.1 continues to apply with the individual factors being the sums of an appropriate number of terms of Equation 3.2. As a final version of this first illustrative example, assume that odd-numbered items are provided with one spare, while the even-numbered items are provided with two spares; then the mission system P_s becomes:

$$P_{s7} = (0.99995)^5 \times (0.9999999)^5$$

$$= 0.99975$$

It was stated at the beginning of the first illustrative example that it represented the least difficult configuration to evaluate. The most frequently occurring complication to this initial simple series configuration is the duplication of items in the logic diagram as shown in Figure 3.3. In this second illustrative example, there are two of each of the ten difficult equipment items. If no sparing capability is provided, the expression for mission system P_s, 3.1A, is merely an extension of the original general expression 3.1.

$$P_{sA} = \prod_{i=1}^{NN} (R_i) = R_A \times R_{AA} \times R_B \times R_{BB} \times \ldots \times R_N \times R_{NN} \qquad (3.1A)$$

Carrying forward the same equipment item performance values, $\lambda_i = 1.0 \times 10^6$ and $t_i = 10^4$ hours, the mission system P_s is:

$$P_{sA} = \prod_{i=1}^{NN} (R_i) = (0.9900)^{20}$$

$$= 0.8181$$

In this example the general Item I and Item II are identical. Therefore, any spares of this type could be used for either item. Specifically, if a single spare for the item A- Item AA combination were provided, the logic diagram would appear as in Figure 3.4. The analytic approach which appears to be most convenient to use in this sort of situation is to replace the Item A-Item AA combination with an equivalent Item AA1 which has the same duty cycle as Item A or Item AA, but has a failure rate, λ_{AA1}, equal to the sum of the failure rates of Items A and AA, or $\lambda_A + \lambda_{AA}$. With the proper substitutions, the estimate of mission system P_s becomes:

$$P_{s7} = R_{AA1} \times \prod_{i=3}^{NN} (R_i) = \left[e^{-\lambda_{AA1} t_{AA1}} \times \lambda_{AA1} t_{AA1} \left(e^{\lambda_{AA1} \; AA1} \right) \right]$$

$$\left(e^{-\lambda_i t_i} \right) = \left[e^{-0.02} + 0.02\, e^{-0.02} \right] \left(e^{-0.01} \right)^{18} = \left[0.82 + 0.1625 \right]$$

$$(0.9900)^{18} = \left[0.9825 \right] (0.8357) = 0.8211$$

Rather than follow through with a numerical computation with each variation of this example as was done in the first example, it seems appropriate to attempt to generalize and, if possible, develop a general expression which can be used to analyze systems with this type of logic diagram and with any combination of sparing provisions. Considering again the general

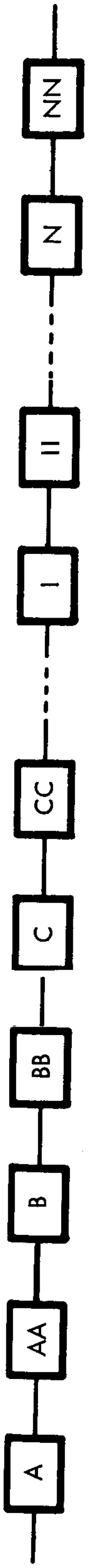


Figure 3.3. Duplication of Items in Logic Diagram

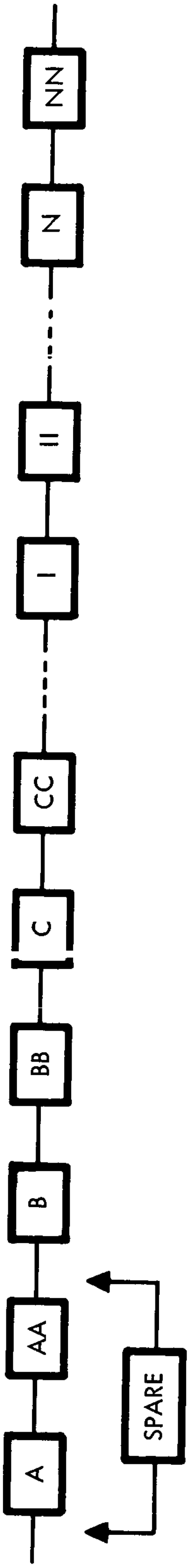


Figure 3.4. Logic Diagram Showing Replacement A-Item, AA Combination

item I of the basic series system diagram, there may be any number 1 through a_i of these identical items operating as a series subset with any number of spares, 0 through b_i, as represented in Figure 3.5. This situation presents a rather complex derivation, based upon an analysis of possible operating combinations and requires the development of Markov probability matrices which results in the general Equation 3.6.

$$R_i = \sum_{j=0}^{b_i} \frac{(a_i \lambda_i t_i)^j}{j!} \left(e^{-a_i \lambda_i t_i} \right) \tag{3.6}$$

The details of this derivation are available in the literature. If this general expression for item I is applied to basic Equation 3.1, it can be stated in terms of a_i, b_i, λ_i, t_i, and N as follows:

$$P_s = \prod_{i=1}^{N} \left[\sum_{j=0}^{b_i} \frac{(a_i \lambda_i t_i)^j}{j!} \left(e^{-a_i \lambda_i t_i} \right) \right] \tag{3.7}$$

There must be a further refinement of Equation 3.7 making it applicable to general item I subsets which normally function with a_i items operating, but which can function at some reduced but still acceptable capacity with as few as c_i items operating. An exact expression for this situation is beyond the present state of the art. However, an approximation which is as good as the data base, is possible if this characteristic of the subset is interpreted in terms of effective spares, d_i. This abstract quantity is:

$$d_i = b_i + (a_i - c_i) = b_i + a_i - c_i$$

The incorporation of this parameter into the general expression adapts it to those subsets which will continue to function for $(a_i - c_i)$ failures after the actual supply of b_i spares have been exhausted. For the purposes of P_s estimates, there are effectively d_i spares. Equation 3.7 modified to include this parameter is

$$P_s = \prod_{i=1}^{N} \sum_{j=0}^{d_i} \frac{(a_i \lambda_i t_i)^j}{j!} \left(e^{-a_i \lambda_i t_i} \right) \tag{3.8}$$

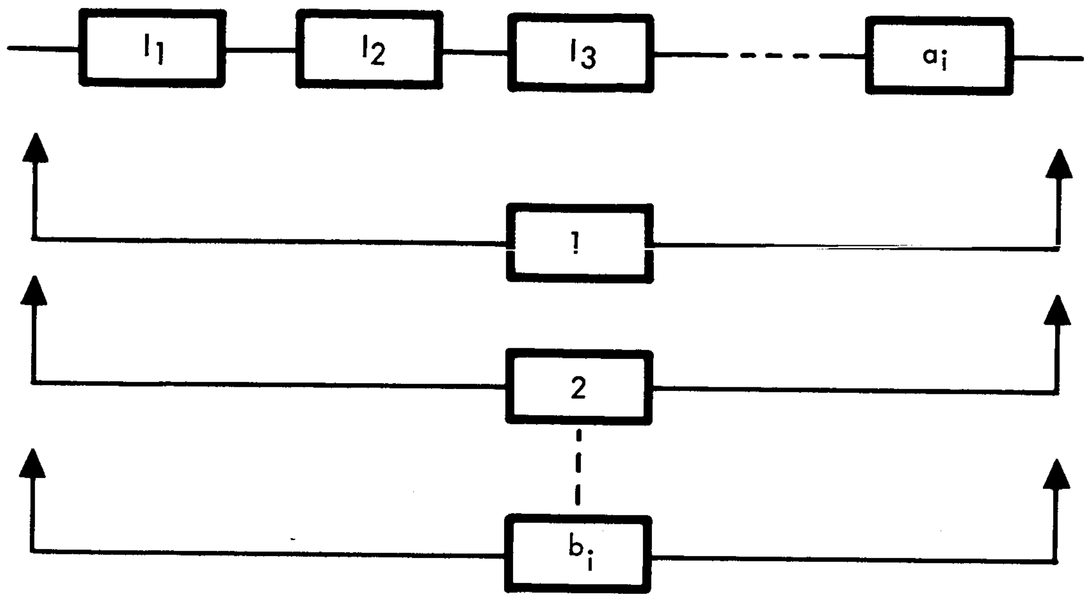


Figure 3.5. Any Number of Identical Items Operating as a Series Subset

The reason that Equation 3.8 is an approximation and not a mathematical exactitude is associated with $(a_i \lambda_i t_i)$ product. After the b_i actual spares have been used this product should be successively decreased to: $[(a_i-1)\lambda_i t_i]$, $[(a_i-2)\lambda_i t_i]$, ... $[(a_i-c_i)\lambda_i t_i]$ as the number of items operating in the subset is being reduced by the (a_i-c_i) failures and the probability of occurrence of additional failures is correspondingly reduced. The amount of error introduced into the P_s estimates by the use of this approximation appears to be negligible. Importantly, the error will always be conservative; that is, the computed P_s estimates will always be slightly lower than a more exact estimate computed with a mathematically exact version of Equation 3.8, yet to be derived.

The concept of a normal operation of a_i items with the possibility of a backup, or minimum operation of c_i items, eliminates the usual distinction between series subsets and parallel subsets when the condition of minimal operation is <u>any</u> c_i items of the normally operating a_i items. In terms of logic diagrams, Figure 3.6 and Figure 3.7 are equivalent under this condition. Such a condition exists when "cannibalism" or the interchange of a failed item with a like operating item in the subset is permitted. This has been found to be the case in a large proportion of the anticipated long duration mission subsystems.

Estimates of safety/success improvements involving the use of Equation 3.8 can quickly become arduous. A graphic aid to facilitate the use of this and similar expressions has been developed under this study and is presented in Figure 3.8. This plot is based upon the Equation:

$$P_s = \sum_{j=0}^{k} \left(\frac{r^j}{j!} e^{-r} \right)$$

This family of curves relates the values of R and r of integer values of j from 0 to 20 on special log coordinates. The correspondence of k to d_i and r to $(a_i \lambda_i t_i)$ is clear.

3.2 EQUALIZING RISK WITHIN A FUNCTION

Given that Equation 3.8 provides a means of assessing the effects of sparing weak links within a subsystem function, it yet remains to identify the individual requirements. That is, it must be determined the required level of reliability for each component within a function which, when achieved, will assure meeting the required contribution to the mission P_s. Ideally, each item in a subsystem, with its provided maintenance capability, should present the same "risk" or probability of causing the subsystem to become

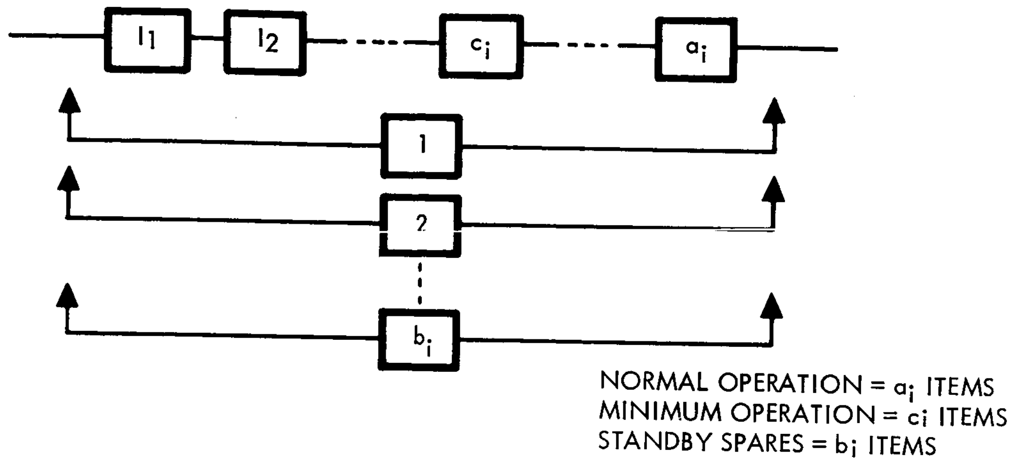


Figure 3.6. Concept of Normal Operation With Backup

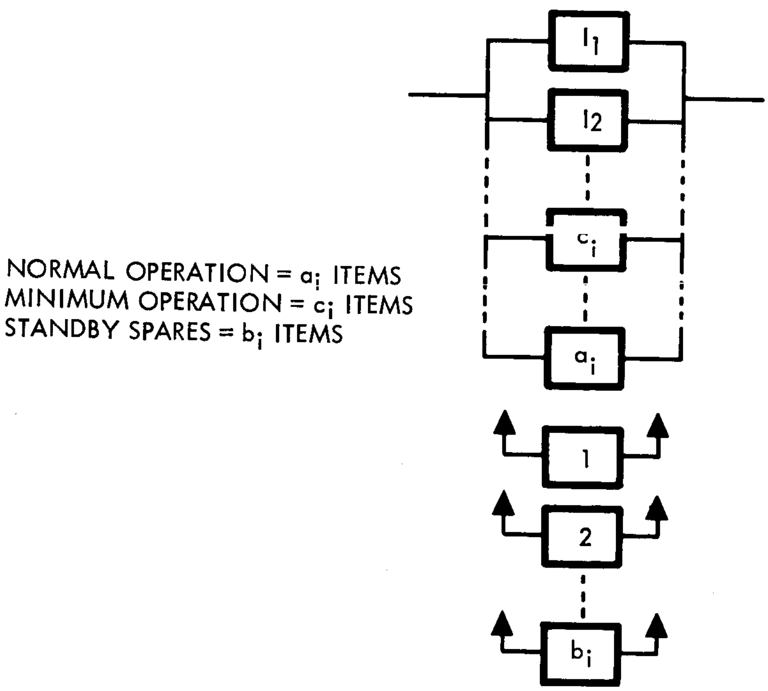


Figure 3.7. Concept When Interchange of
Failed Item With Similar Item

non-functional by virtue of its lack of supporting spares. The "equalized risk" concept was developed in an attempt to provide a means to establish a reasonable reliability requirement for components at a given level, with equal risk within the criticality class.

To illustrate, consider the example subsystem of four different kinds of items, A, B, C and D, shown in Figure 3.9. This configuration, in which item A appears twice and item B three times, will have a subsystem contribution to P_s, that is a function of the reliabilities of the four items as shown in Equation 3.9.

$$P_s = R_A \times R_B \left\{1 - \left[1 - R_A \times R_C \times R_D\right] \left[1 - (1 - R_B)^2\right]\right\} \tag{3.9}$$

$$= R_A R_B - 2R_A(R_B)^2 + 2(R_A)^2 R_C R_D - R_A(R_B)^3$$

$$+ (R_A)^2 (R_B)^3 R_C R_D$$

If the concept of equalized risk is applied, the reliabilities of items will be made equal by design and maintenance capability provided.

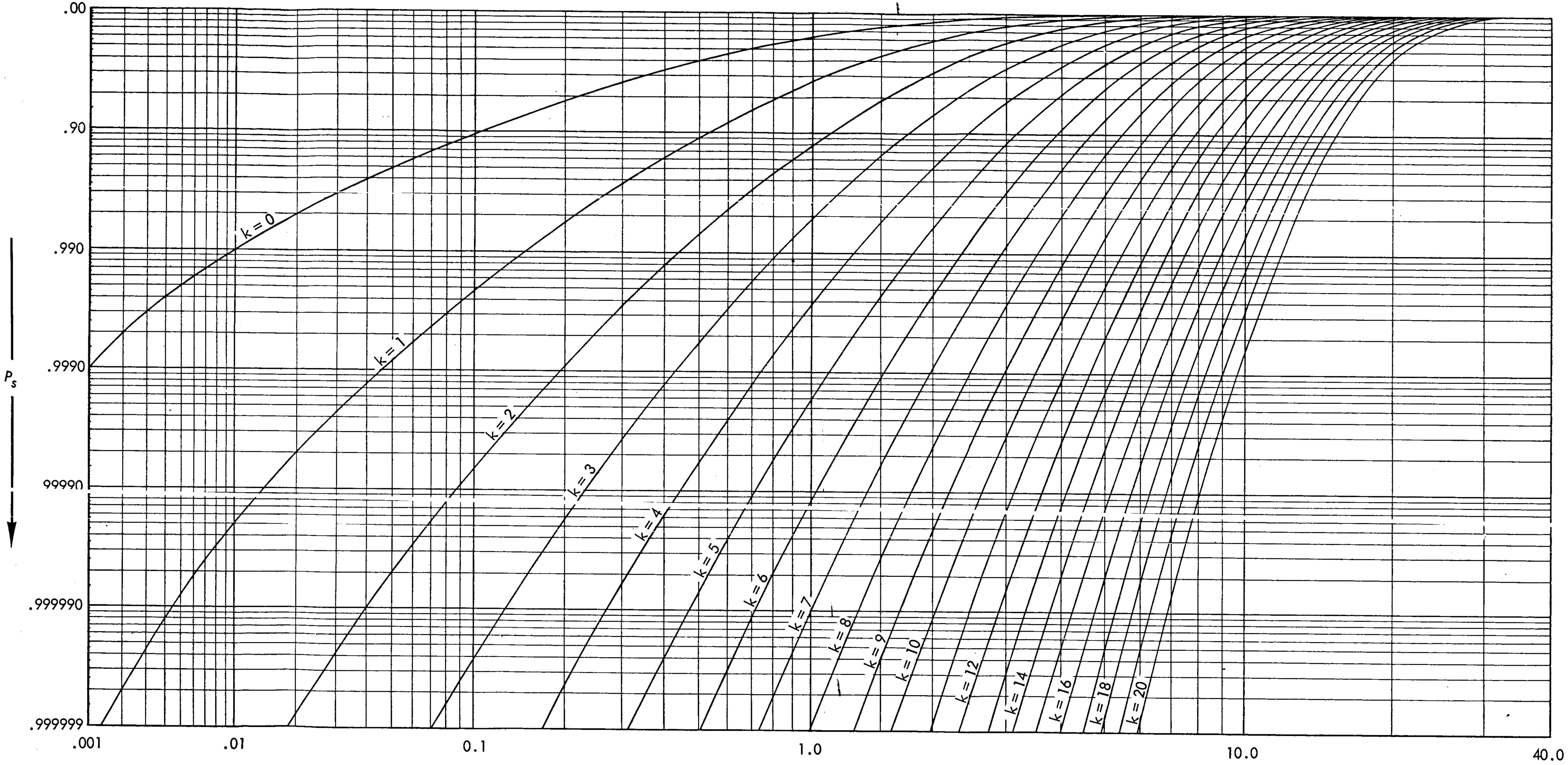
$$R_i = R_A = R_B = R_C = R_D$$

In which case, Equation 3.9 reduces to expression 3.10.

$$P_s = (R_i)^2 - 2(R_i)^3 - (R_i)^4 + 2(R_i)^6 + (R_i)^7 \tag{3.10}$$

For a given subsystem P_s of: $P_s = 0.8636$, the item contributions must be $R_i = 0.9870$ or better.

To determine the maintenance requirements involved in providing this item a reliability of $R_i = 0.9870$, some maintenance logic diagrams of the type shown in Figure 3.10 are helpful. Care must be taken to assure that such a diagram accurately represents the operational limitations and reliability characteristics of the subsystem. In the case of this example, the assumptions upon which Equation 3.9 is based require that each application of each item, that is, both A items and all three B items, as well as the single C and D items, must be provided with maintenance capability to bring them individually up to the allocated $R_i = 0.9870$ level. Figure 3.10 reflects this in that all seven item applications are shown. It is possible to make



$$P_s = \sum_{j=0}^{k} \left(\frac{r^j}{j!} e^{-r} \right)$$

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Figure 3.8. Graphic Representation Using Equation Estimating Safety/Success

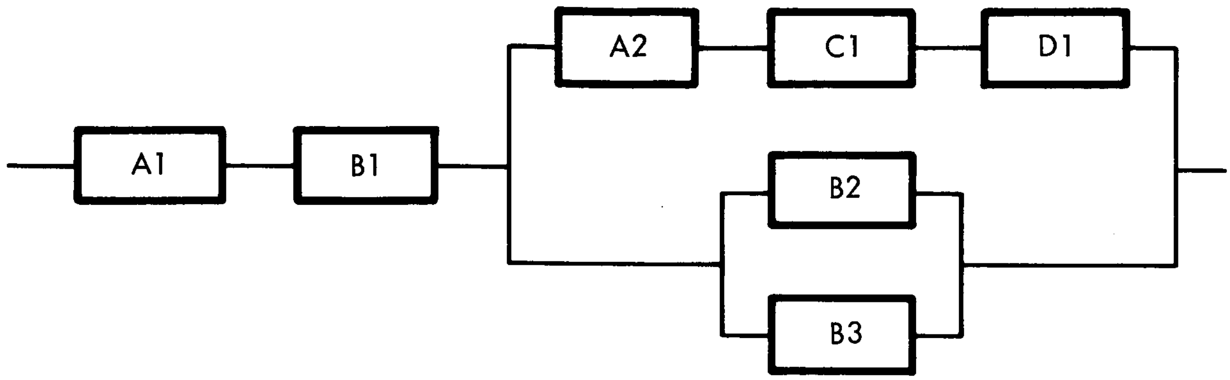


Figure 3.9. Example Subsystem of Four Different Items, A, B, C, D

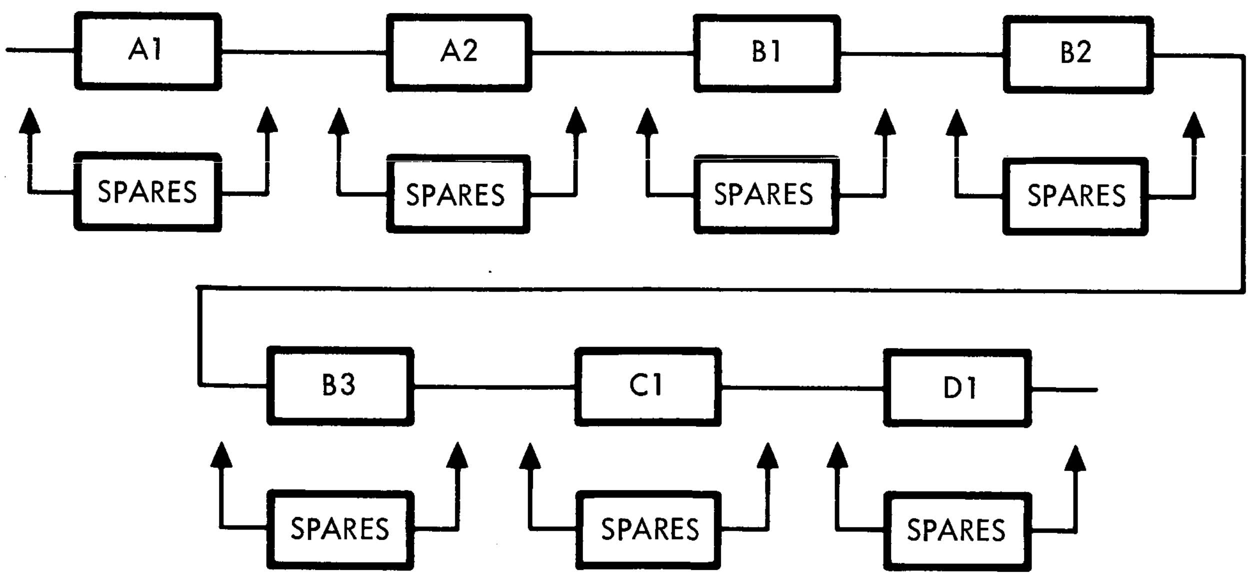


Figure 3.10. Maintenance Logic Diagram

some simplification of this diagram by the following reasoning. Since item A1 and item A2 are physically identical by definition their failure rates are identical, λ_A. Their applications may not necessarily be identical so their duty cycles, or operating times t_{A1} and t_{A2}, must be assumed to be different. If they are physically identical, any A type spare provided may be used in either application and only one store of A type spares need be provided as indicated in Figure 3.11. It is most convenient to determine A type spares requirement if the item A1-A2 combination can be replaced by some analytically equivalent item AA. Item AA will be equivalent to the A1-A2 combination only if it exhibits the same failure probability of occurrence characteristics. This will be true if the below relationship exists.

$$\lambda_{AA}t_{AA} = \lambda_{A1}t_{A1} + \lambda_{A2}t_{A2}$$

Failure rates of identical item and their spares must be equal, therefore

$$\lambda_{AA}t_{AA} = \lambda_{A}(t_{A1} + t_{A2})$$

by a similar line of reasoning,

$$\lambda_{BB}t_{BB} = \lambda_{B}(t_{B1} + t_{B2} + t_{B3})$$

From the above, Figure 3-10 may be simplified to Figure 3.12. One of the purposes of this example was to illustrate how a purely series maintenance logic diagram such as Figure 3.12 may be developed to correspond to the series — parallel reliability logic diagram of Figure 3.9. Once a purely series logic has been developed, Equation 3.8 and Figure 3.8 may be employed to determine the spares requirements.

To carry this example to completion, let the following parameter values be assumed:

Item	Failure Rate	Duty Cycle
A1	19.6×10^{-6}	10^4
A2	19.6×10^{-6}	10^2
B1	30.0×10^{-6}	10^4
B2	30.0×10^{-6}	10

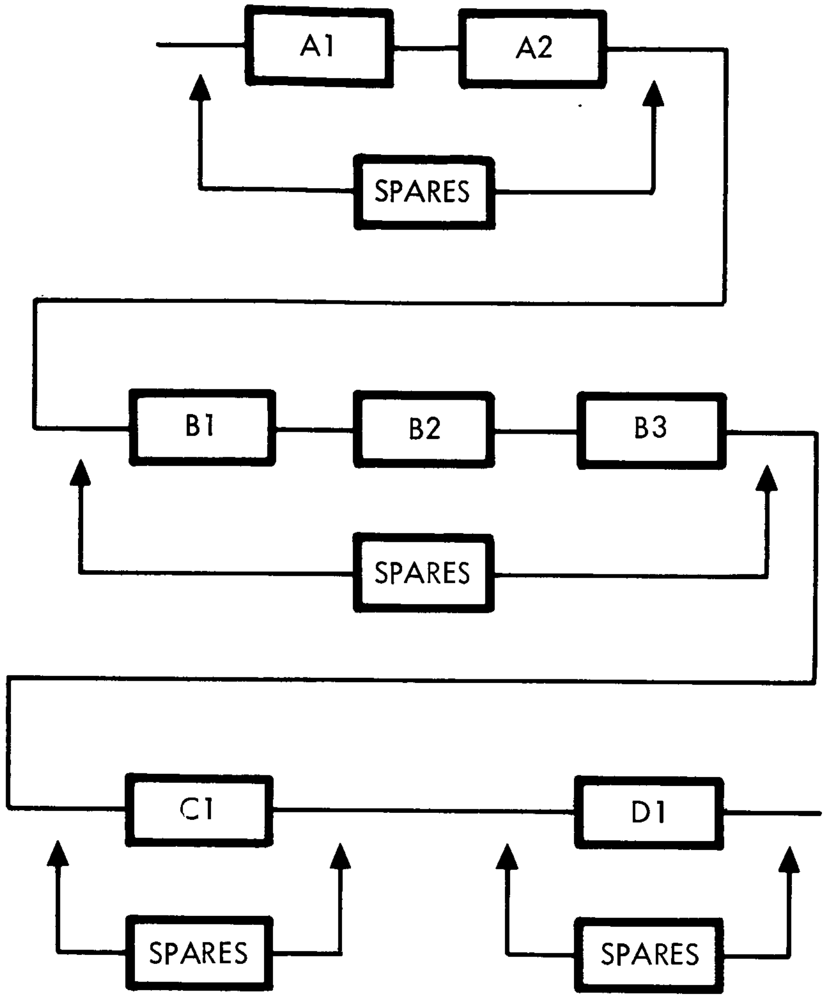


Figure 3.11. Use of an A Type Spare

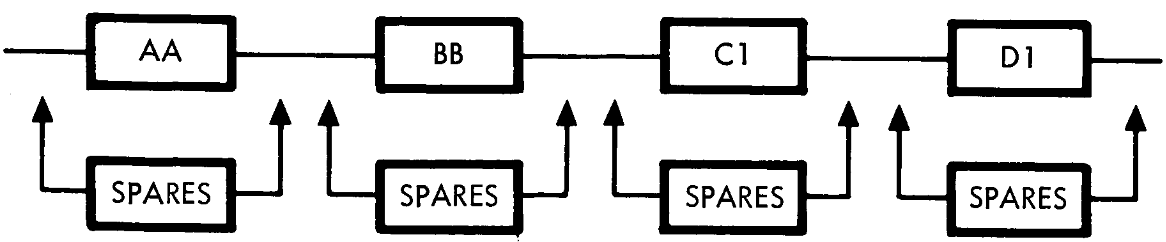


Figure 3.12. Series Maintenance Logic Diagram

Item	Failure Rate	Duty Cycle
B3	30.0×10^{-6}	10
C	90.0×10^{-6}	10^2
D	80.0×10^{-6}	10^2

Interpreted in terms of Figure 3.13, this data becomes

Item	λ	t	λt
AA	19.6×10^{-6}	10,200	0.200
BB	30.0×10^{-6}	10,020	0.300
C	90.0×10^{-6}	100	0.009
D	80.0×10^{-6}	100	0.008

The allocated reliability for all was $R_i = 0.9870$, which can be represented by the dash horizontal line in Figure 3.13, which is a reproduction of Figure 3.8, and is to be used as an aid in the solution of this example. The λt products for each item are plotted as vertical lines in Figure 3.13 and are appropriately labeled. Inspection of this plot reveals that the vertical item AA line and the horizontal allocated reliability intersect between the one-spare-provided and the two-spares-provided curves. The intersection the vertical item AA line with the one-spare-provided curve occurs at a reliability level of about 0.9989. Since it is not possible to provide a fraction of a spare, a choice between providing one spare to the item A1-A2 combination for a reliability slightly less than the allocated level, 0.9823 vs. 0.9870; or providing two spares to the combination for a considerably higher reliability of 0.9987 must be made. If the same procedure be followed for the remaining items and the results summarized in a tabular form.

Item Combination	Spares Req't. Bracket	Resulting Contribution To P_s
A1-A2	1 or 2	0.9823 or 0.9989
B1-B2-B3	1 or 2	0.965 or 0.9967
C	0 or 1	0.914 or 0.9963
D	0 or 1	0.924 or 0.9969

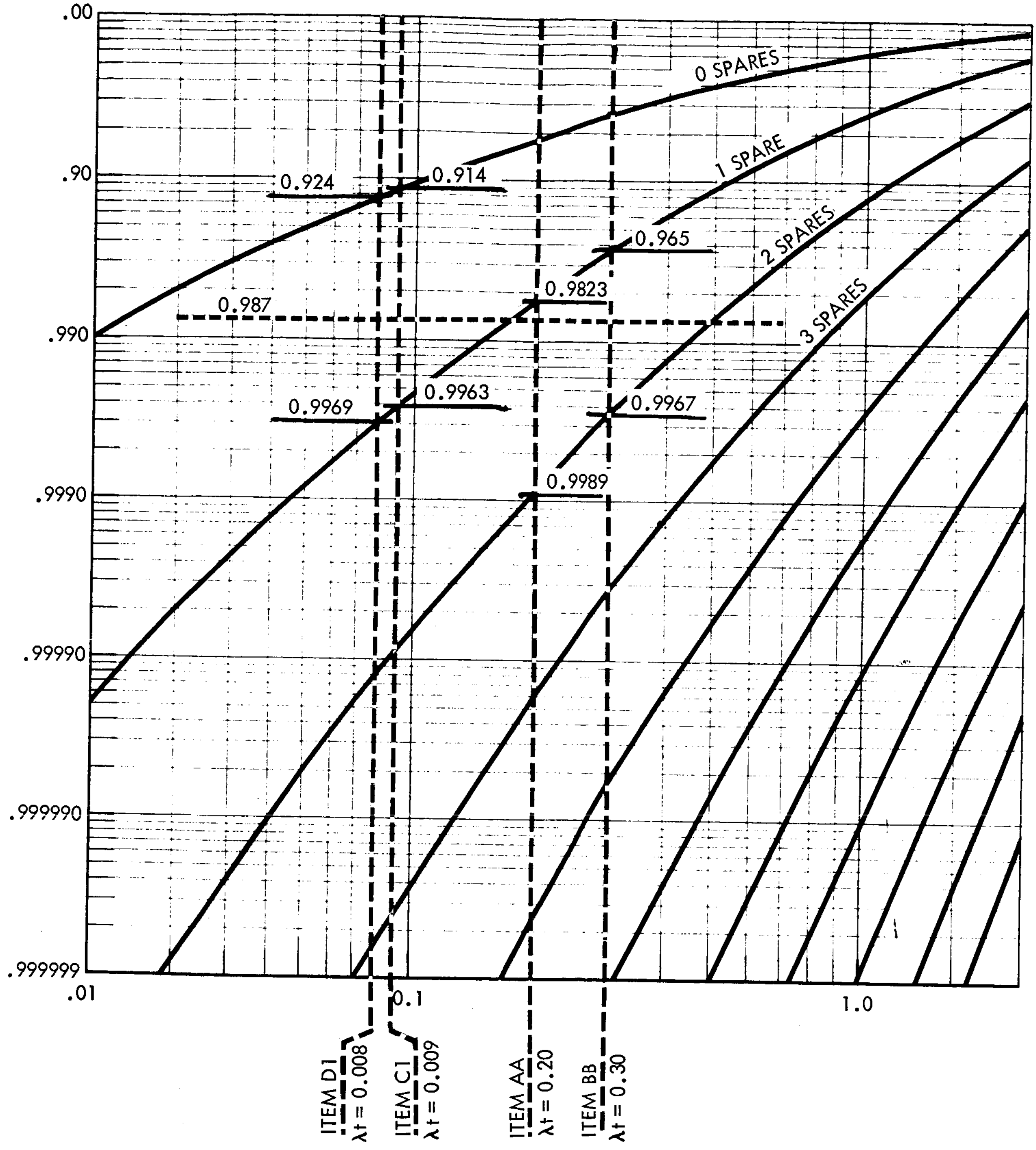


Figure 3.13. A Series to Determine Spares Requirements

This section by no means represents a complete investigation into the problems of P_S estimation for systems requiring maintenance. The intent is to establish a general case starting point upon which the specific subsystem cases in later sections may be based and to augment the available literature where weaknesses precluded application to this study.

IV. CREW CRITICAL (CRITICALITY I) SUBSYSTEMS ANALYSES

The subsystems functions required to assure crew safe return, Criticality I, are analyzed individually in the subsequent paragraphs. The availability analysis has been applied to those required to lower the individual subsystem contribution to the probability of safe return (P_{CS}) to 0.999 or less.

4.1 THE ENVIRONMENTAL CONTROL AND LIFE SUPPORT SYSTEMS

Note: The information used herein was derived through NAR/SD in-house studies performed under the basic technology research program and conducted with the use of contributions from AiResearch Division of Garrett Corporation in Reference 4.1 and 4.2.

A detailed analysis of the baseline mission Environmental Control and Life Support System ECLSS is presented in this section of the report to demonstrate application of the availability concept on a system design, and to demonstrate the scope and effectiveness of the analysis. The baseline system used herein was developed as a part of the system engineering activity and as a necessary prerequisite to the study objectives. It represents the most detailed design yet accomplished of an ECLSS for extended missions. The hardware was derived from Apollo components and other contemporary space programs. Very few new items of hardware were required and none are considered marginal in relation to the state-of-the-art; most have already been demonstrated as reliable processes.

4.1.1 The ECLSS Functional Requirements

Basically, the ECLSS must support both men and equipment for the 700-day baseline mission by providing the required environmental characteristics. It must provide the four-man crew with an atmosphere controlled in terms of both content, temperature and pressure for both the mission module (MM) and the earth entry module (EEM). It must provide temperature control for all temperature sensitive systems in both modules.

In addition, the mission spacecraft will have a probe compartment (PC) which will house the planetary probes. No EC/LSS functions are expected to be provided to this compartment. Temperature control may be a future requirement but that should have a negligible effect on this study since they are Criticality III. The EEM will be occupied by the crew only during ascent, docking and velocity changes. The PC will be entered for a maximum sporadic occupancy of about two weeks for the checkout of scientific equipment. For the purposes of total mission availability support requirements determination, the MM may be considered to be the module requiring a detailed analysis of the EC/LSS support requirements. The system must provide its functions for the full 16800 hours of flight time, with off time only for maintenance and repair.

The MM is itself divided into two main compartments or equipment floor levels. Equipment floor A is the primary crew compartment having maximum radiation and meteroid protection, and the one containing the displays and controls necessary for the operation of the EC/LSS. Equipment floor B contains approximately half of the EC/LSS equipment items which, to some degree, may be considered the "slave" half that backs up the "master" half if equipment floor A. However, this is not a standby backup relationship. Generally, both sets of equipment will be in operation at less than full capacity and sharing equally in the total load. The system layout diagram is presented in Figure 4. 1. It illustrates the concept of two equipment floors, each containing approximately half the EC/LSS equipment.

There are six fundamental subsystem functions with the basic interrelationships which are shown in Figure 4. 2. Because of the interrelationships and equipment operational interdependencies, these functions have been arbitrarily divided into thirteen EC/LSS functional equipment groups as identified in Figure 4. 3. These have been further expanded into a total subsystem flow schematic showing the individual items of equipment in Figure 4. 4, which forms the basis of subsequent analysis. These functions work together as related in Figure 4. 1 and 4. 2, as follows:

Four equipment groups are necessary for the spacecraft thermal control function. Of these, equipment group 1. 0 is the radiator circuit whose functional purpose is the rejection of unwanted heat energy into space by pure radiation. This group is thermodynamically connected to equipment group 2. 0, the refrigerant circuit, by means of a refrigerant fluid such as Freon. The functional purpose of group 2. 0 is to accept unwanted heat energy from the atmosphere circuit, group 3. 0, and the equipment coolant circuit, group 4. 0, and transfer it to the refrigerant fluid from which it will be rejected to space. It may also transfer heat from the coolant circuit to the atmosphere circuit if the crew compartments require heating rather than cooling. The cabins are insulated for a thermal balance so that with normal equipment loads the temperature is as required. The purpose of group 3. 0, the atmosphere circuit, will be the circulation of a breatheable atmosphere through the crew compartments and a set of refrigerant circuit heat exchangers. Group 4. 0, the coolant circuit, performs a comparable purpose in the circulation of a collant fluid through the condensers, evaporators, cold plates, etc., of other equipment groups from which unwanted heat energy must be removed and finally through another set of refrigerant circuit heat exchangers. The major subsystem functions are as follows.

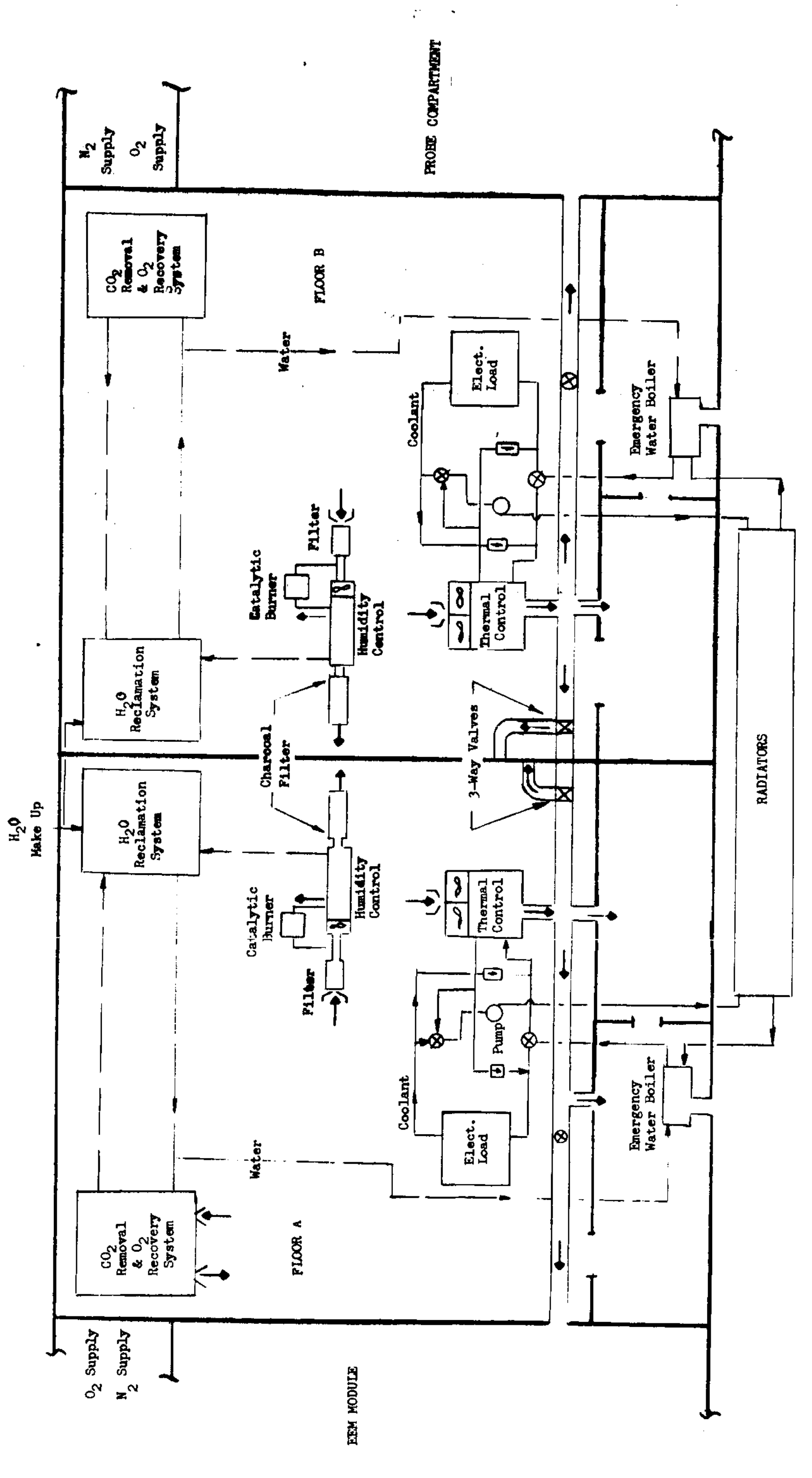


Figure 4-1. - System Layout Diagram, Showing Concept of Two Equipment Floors

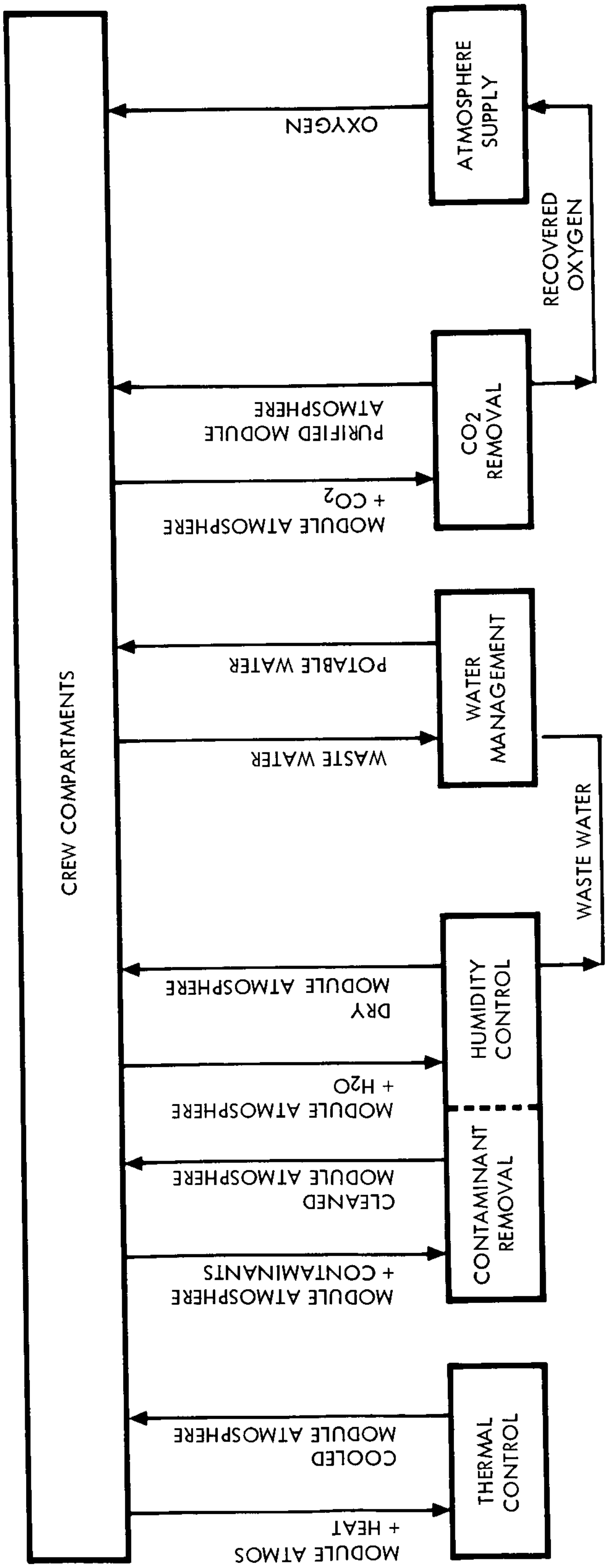


Figure 4.2. Six Fundamental Subsystem Functions With Basic Interrelationships

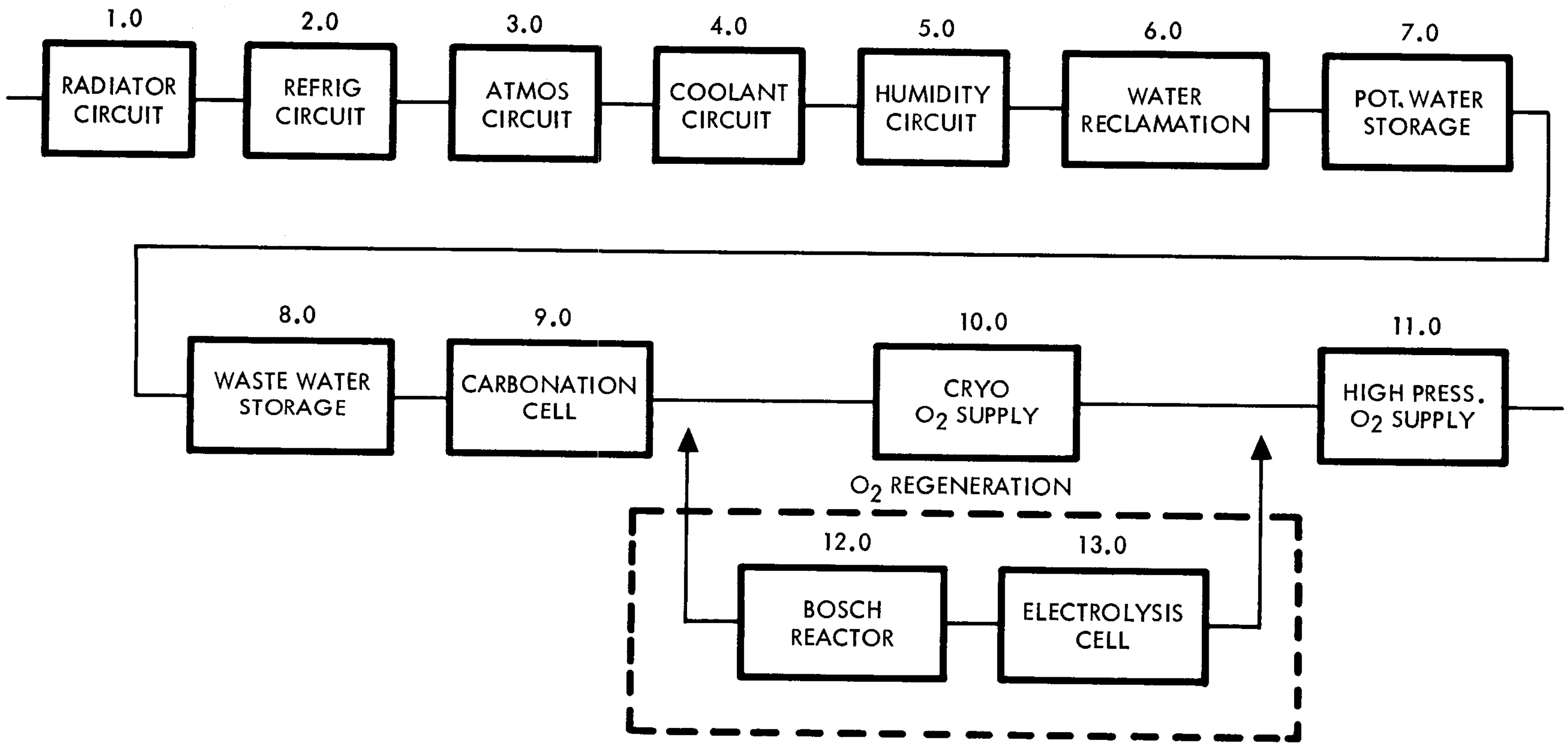


Figure 4.3. Graphic Presentation of 13 EC/LSS Functional Equipment Groups

1. The functions of humidity control and contaminant removal are performed by one equipment group, the humidity circuit, 5.0. Excess humidity is removed from the crew compartment atmosphere and transferred as waste water to waste-water storage. Harmful contaminants will then be removed from the atmosphere by means of a series of filters and a catalytic burner function.

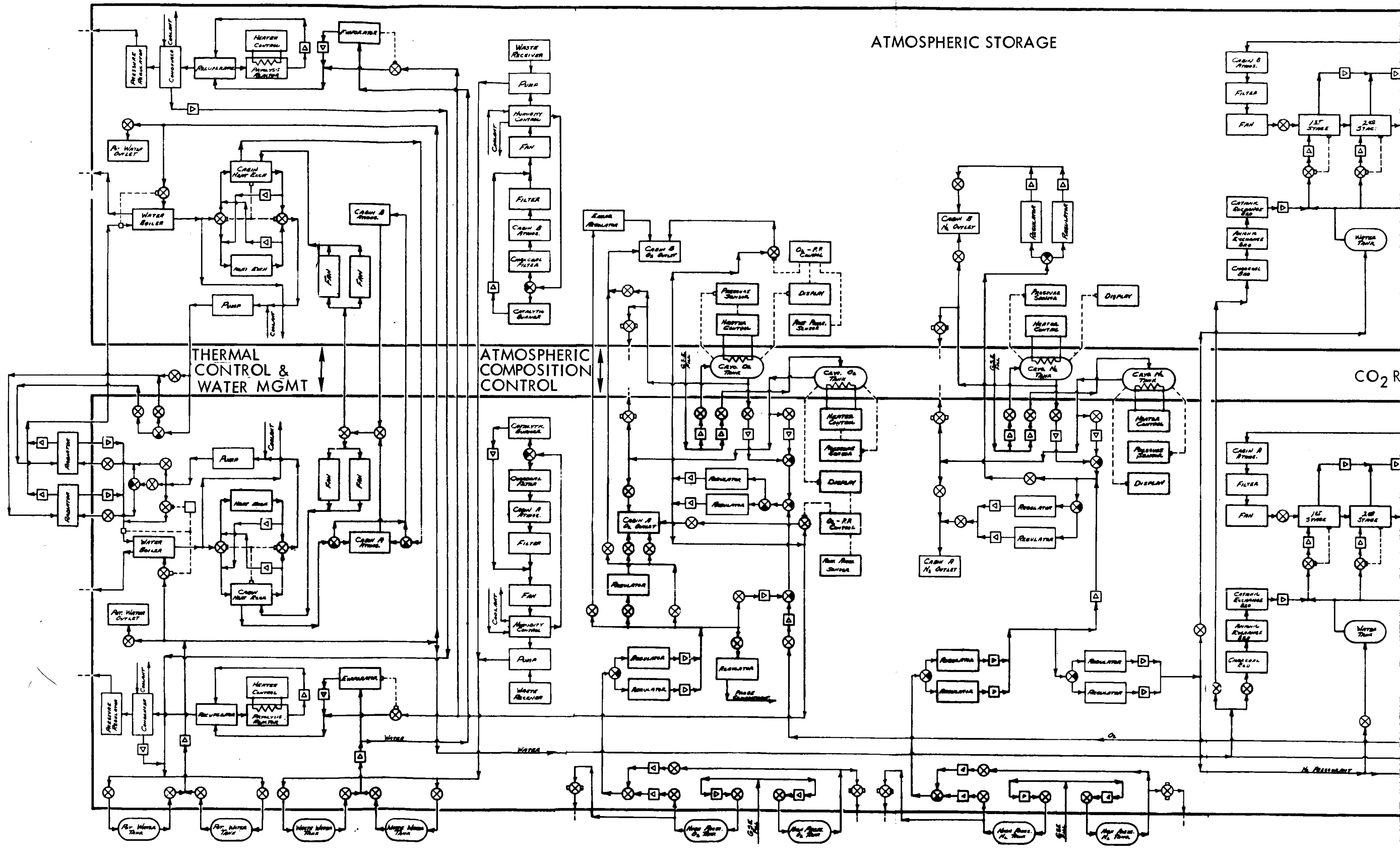
2. The water management function is performed by three equipment groups; the water reclamation group, 6.0, the potable water storage group, 7.0, and the waste water storage group, 8.0. The purposes of these groups are clearly implied by their titles. The waste-water storage group stores waste water until enough has been accumulated to make up a batch which is then reclaimed by the water reclamation group. The pure water is then stored by the potable water storage group until needed. A detailed description of the reclamation process is included.

This system recovers water by evaporating it from waste water and urine at 100 F. Each man will require 9.75 pounds of water per day for drinking and food preparation. There will be a requirement for makeup water of 1.27 pound per man, per day. For a four-man crew and 700-day mission, the water weight at launch must exceed 3550 pounds with a water recovery system. Without it, it must exceed 27,300 pounds. Obviously, a water recovery system is essential. The urine vapor with oxygen added is passed through a high-temperature catalyst bed to oxidize trace contaminants.

Evaporation is controlled at 100 F and 0.6 psi in the evaporator to minimize hydrolysis of urea in the urine. The water vapor and added oxygen are heated in the recuperator by the vapor leaving the catalyst bed which bed will be heated by an electrical heater. In the catalyst bed, the oxides, and CO_2 are removed. The product flows through the recuperator into the condenser, where water is condensed and removed. The trace gases are periodically vented to vacuum.

The recuperator, heat source, and catalyst are combined into a single integrated unit. The evaporator is designed as a plastic bag with bonded absorbent surfaces for wicking and evaporation. When the wicking becomes saturated with salts, the bag is replaced.

3. The removal of carbon dioxide from the atmosphere is to be accomplished by the carbonation cell, group 9.0. Under normal operation CO_2 removed from the atmosphere will be dumped overboard into space. In the event of loss of access to the O_2 source or an inadequate remaining supply, the



FOLDOUT FRAME 1

FOLDOUT FRAME 2

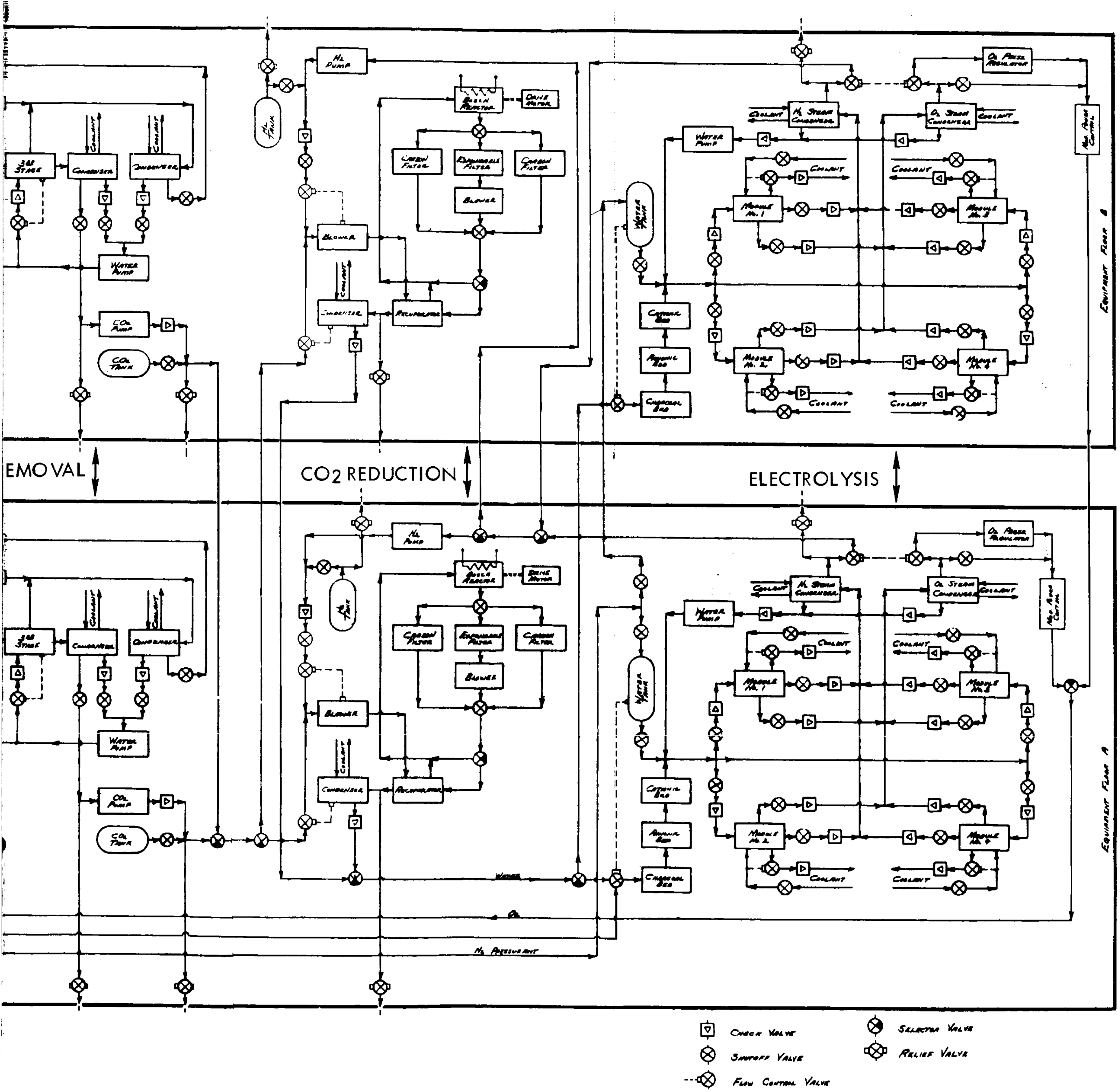
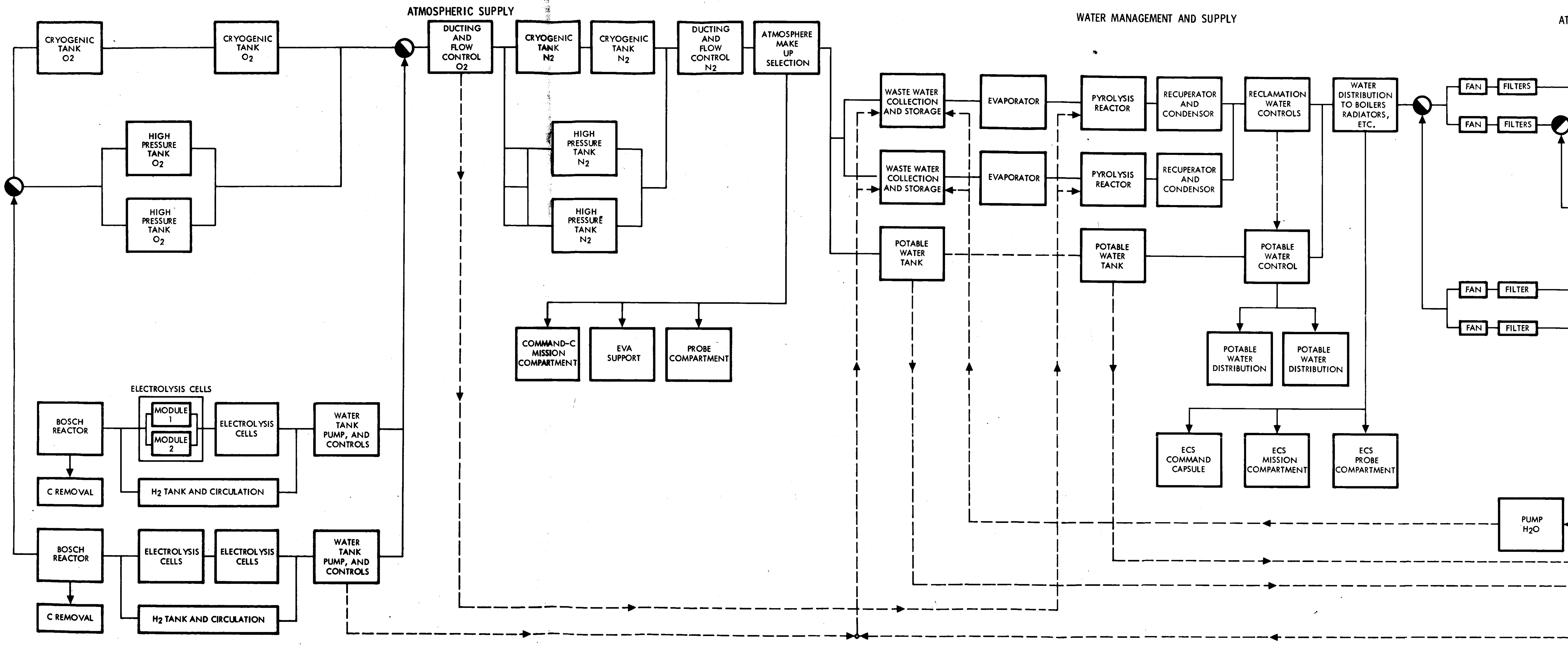


Figure 4.4. Preliminary Schematic of Fully Integrated EC/LSS



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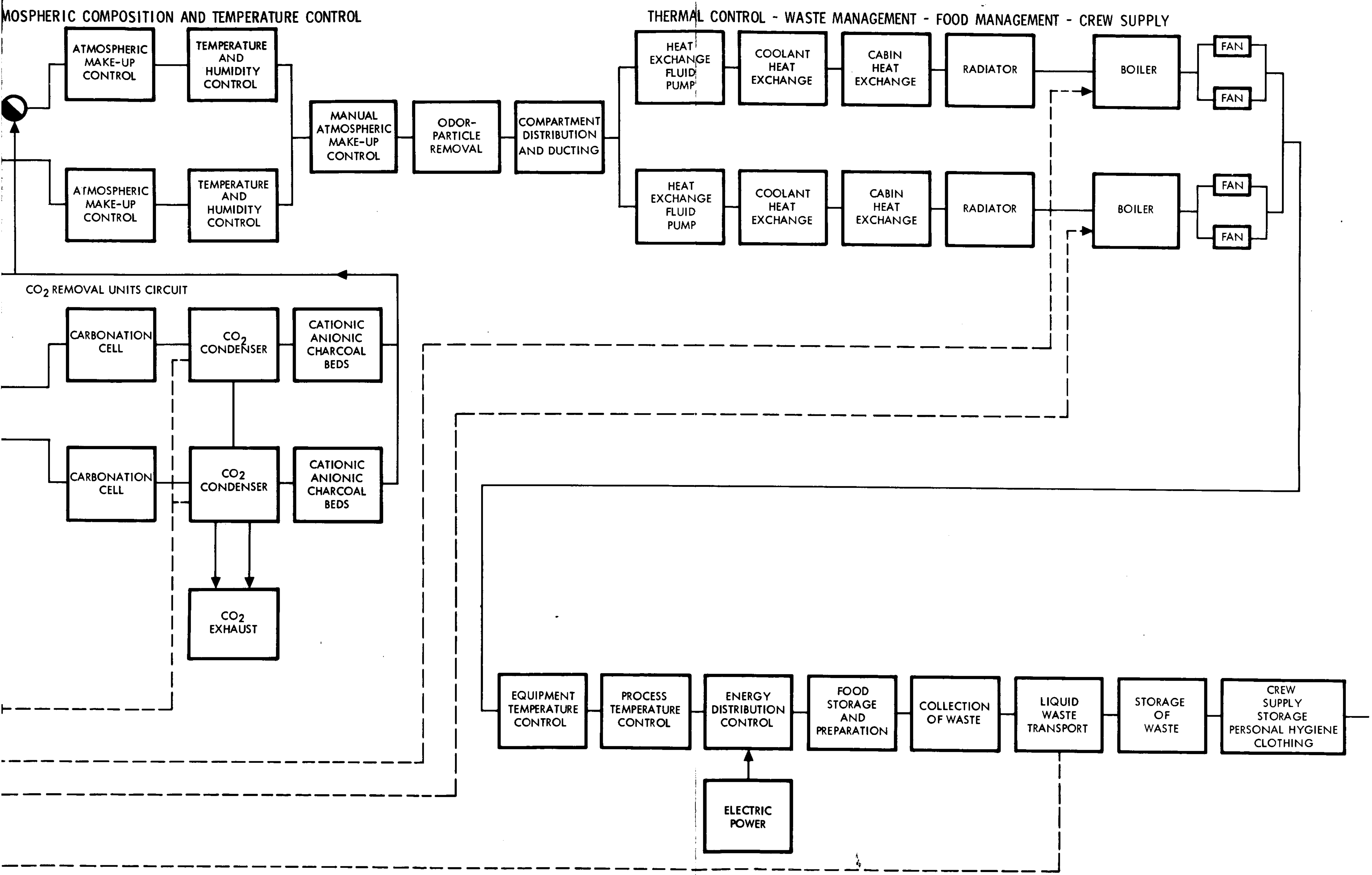


FOLDOUT FRAME 4

Figure 4.5. Environmental Control and Life Support Subsystem, Logic Diagram

FOLDOUT FRAME 5

cabin CO_2 will then become an input to the oxygen regeneration function. A description is included herein because it is a new process, not described in the references.

The carbonation cell, CO_2 concentrator, is an electrochemical device capable of removing CO_2 from cabin atmosphere. When selected electrodes and electrolytes are combined in an electrochemical cell and a voltage potential is impressed on the cell, concentration gradients and migration of ionic species are induced at the electrodes. The CO_2 concentration setup consists of three cells. In the first stage, CO_2 and oxygen are removed from the cabin process gas and transferred from cathode to anode through the electrolyte K_2CO_3 as carbonate, bicarbonate, and hydroxyl ions. At the anode, CO_2 and oxygen are reformed and discharged. This gas mixture, substantially enriched in CO_2, enters the second stage. There the process is repeated. However, because of the CO_2-rich feed gas, most of the CO_2 transfer occurs through formation of the bicarbonate ion and results in transfer of proportionally more CO_2 than oxygen. The second stage gas enters the third stage for final processing. The third stage differs from the first two stages in that is selectively removes oxygen from the feed gas to produce a mixture of greater than 90 percent CO_2, ready for further processing or disposal.

The system is cooled by evaporation of water and by the process gas stream. Power requirements have been estimated as 157 watts per man. It is a continuous system, having minimal moving parts (blowers, fan) and is simple in operation.

4. Atmosphere supply, will be accomplished by two equipment groups: cryogenic oxygen supply, 10.0, and high-pressure oxygen supply, 11.0. The basic source of supply is the cryogenically stored oxygen with the high-pressure oxygen supply being necessary only to provide emergency repressurization capability which the relatively slow gas flow characteristics of cryogenic supply does not provide. Included as part of the cryogenic oxygen supply equipment group, 10.0, are the sensors, regulators, etc., necessary for the proper oxygen pressurization of the crew compartments.

As a partial backup to the cryogenic oxygen storage, a Bosch reactor group, 12.0, and an electrolysis cell group, 13.0, are included to recover metabolic oxygen from atmosphere CO_2

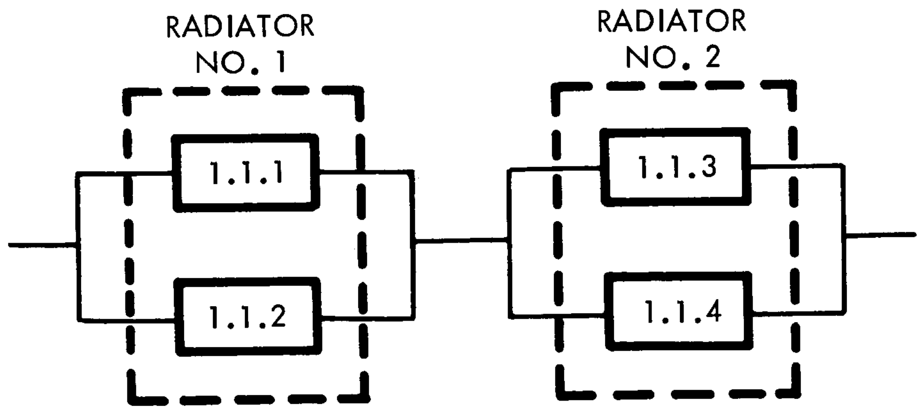


Figure 4.6. Two Radiators, Two Tubes Each

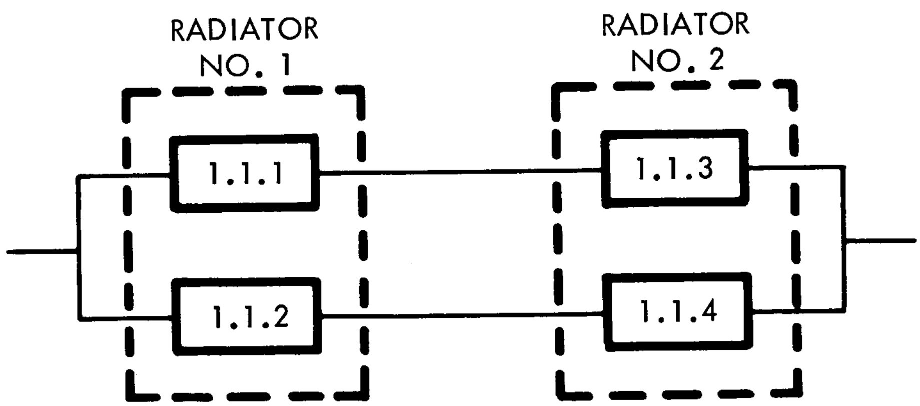


Figure 4.7. Radiator Tube Logic

exchangers, etc., are in full operation. Therefore, the operating relationships, as depicted by a reliability logic diagram, are best represented by Figure 4.7 for these tube sets.

The remaining three equipment items in the radiator circuit group have no unusual characteristics or operational constraints and will operate, one each in series, with any of the radiator tube sets. Since these are interchangeable, one form of maintenance action which can be performed is the interchange of a failed item with a like item operating elsewhere in the group (cannibalism) to reestablish a functional combination within the group. The resultant reliability logic diagram from the radiator circuit with items exhibiting the described characteristics is shown in Figure 4.8.

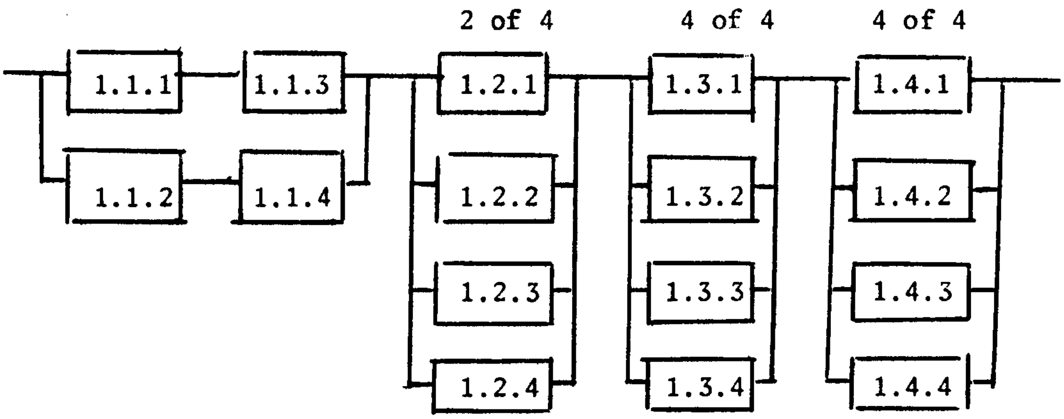


Figure 4.8. Radiator Circuit Logic

Applying the values in Table 4.1:

$$P_s\ (1.1) = 1 - 1-\left(e^{-\lambda_i t_i}\right)^{2\ 2}$$

$$= 1 - 1-(e^{-.00168})^{2\,2}$$

$$P_s\ (1.1) = 0.999989$$

which appears in column 9 of the Table.

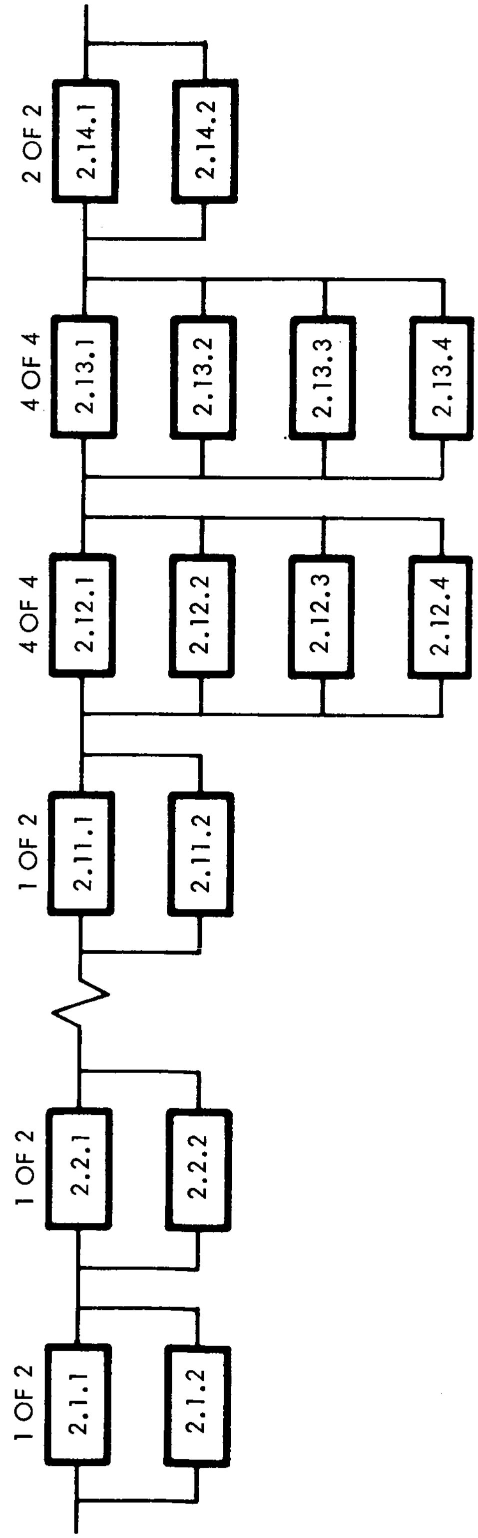


Figure 4.9. Refrigerant Circuit Logic

The recommended contribution to P_s for this system should be not less than 0.999 since this is the most crew sensitive system. To meet the objective all of the 13 subsystems must be improved considerably. Since each subsystem is, or contains weak links, all were evaluated individually in an availability analysis.

4.1.3 Availability Analysis, ECLSS

To decrease the failure hazard to the acceptable level 1×10^{-3}, and to achieve the equalized risk objectives established for this study, the allowable unreliability should be allocated as in Section 3.2 of this report. However, an effective approximation can be made by counting the number of sparing possibilities within the ECLSS and allocating the requirement over this number of instances in which maintenance capability may be provided. The results can then be checked after the sparing or maintenance requirements analysis has been completed, then adjustments made when required.

The functional groups of the eleven ECLSS subsystems were counted and found to be about 85 of approximately equal complexity. Each represented a weak-link possibility, and therefore, a candidate for an M and R action. All were not realistically sparing possibilities. For instance, the radiator tube sets and external storage tanks are known to be inaccessible for repair or replacement. However, let the "first cut" goal be taken as the 85 root of 0.9990 and the resulting first cut maintenance requirements analysis be evaluated and modified as necessary. The 85th root of 0.9990 is 0.9999882, or 11 failures in 10^6 missions which will be taken as the equal risk reliability level desired for each sparable weak link.

Each of the eleven subsystems or groups were then analyzed for requirements to meet or surpass the required P_s. The results are described in the following paragraphs.

1. Radiator Circuit. There are four different kinds of equipment items in the radiator circuit, 1.0, as indicated in the first two columns of Table 4.1. This group includes four space radiator tube sets which are potentially weak links, physically contained in two radiators of two tube sets each, as diagrammed in Figure 4.6. Any two tube sets, not in the same radiator, are required to dissipate the expected normal heat load. On this basis, the four acceptable operating combinations of radiator tube sets would be: 1.1.1 with 1.1.3, 1.1.1 with 1.1.4, 1.1.2 with 1.1.3, and 1.1.2 with 1.1.4. A further operating limitation exists in that the 1.1.1 with the 1.1.4 and the 1.1.2 with 1.1.3 combinations may be used only if the refrigerant circuit is operating in the primary mode, i.e., both pumps, all four heat

4.1.2 Reliability Assessment, EC/LSS

For the preponderance of the ECLSS, the requirements associated crew safe return, P_s, or Criticality I, include most of the functions. The system reliability logic diagram is presented in Figure 4-5.

A reliability estimate of the system was made using the Apollo failure rate data as reflected in References 4.1-2 and 4.1-3. The results are as reflected in Table 4.0; the system reliability is estimated to be only 0.104. This means that only a 10-percent chance exists that there will not be a loss of some critical function. Obviously, the risk is too high and the weak links must be corrected.

Table 4.0. EC/LSS Reliability Estimates, Subsystem Level

Logic Code	Subsystems	Duty Cycle (hours)	Estimated Reliability
1.0	Radiator circuit	16,800	0.9316
2.0	Refrigerant circuit	16,800	0.7756
3.0	Atmospheric circuit	16,800	0.6149
4.0	Coolant circuit	16,800	0.8658
5.0	Humidity circuit	16,800	0.9243
6.0	Water reclamation	16,800	0.8929
7.0	Potable water	16,800	0.9838
8.0	Waste water	16,800	0.9773
9.0	Carbonation cell	16,800	0.8612
10.0	Cryogenic supply	16,800	0.4583
11.0	High-pressure O_2	16,800	0.8560
12.0	Bosch reactor*	1680	0.9174
13.0	Electrolysis*	1680	0.9809
System			0.1041

*Required only in case of loss of some cryogenic storage capability collected by the carbonation cells. Leakage O_2 must still be made up from one of the forms of storage.

Metabolic oxygen can be cycled and recycled for reuse. During the metabolic process, oxygen is consumed by the body and then discharged in solid, liquid, and gaseous wastes. The greatest part of this oxygen is used in oxidation of organic components and is expired as carbon dioxide, with most of the remainder appearing as metabolic water. Of the numerous possible means by which oxygen may be recovered from CO_2, the Bosch reaction system was selected as candidate for Mars flyby application. It is based on the principle of hydrogeneration of CO_2 to C and H_2O, followed by electrolysis of the product water to obtain oxygen.

The reduction of CO_2 with hydrogen to carbon and water in a one-step process has been investigated, and engineering models have been built and tested. Investigators have found that carbon monoxide and methane are formed in the reactor over iron catalysts, and that the presence of methane is necessary to improve efficiency. One-cycle conversion is of low efficiency; hence, recycling of the product gas is required. The rate of water formation is sensitive to the presence of water in the inlet stream, so the product gases must be dried to low dew points before recycling. System power requirements are intimately related to reactor size, insulation, and effectiveness of the thermal generation process between the cool reactants and the hot product gases.

5. The nitrogen supply is a desirable item but not truly necessary to crew safe return. Under normal conditions, a two-gas atmosphere of about 50/50 oxygen and nitrogen will be supplied to the crew compartments. If the nitrogen supply becomes non-functional, a pure oxygen atmosphere at a reduced pressure will continue to provide a safe environment from the oxygen supply. Since the function of nitrogen supply is not considered to be critical to crew survival and therefore will not have a significant effect upon the P_s, it could be considered a Criticality II function and impose no decrements on the assessment of crew safe return.

Table 4. 1. Radiator Circuit Analysis

Logic Block Number	Component	Duty Cycle (hours)	Failure Rate x 10^6	Reliability	Number of Spares	Contribution to P_s	Spares Weight (-lbs)
1.1, 1-4	Radiator tube set	16,800	0.10	0.999989	0	0.999989	0
1.2, 1-4	Radiator flow control		10.0 *	0.97	4	0.999993	8.0
1.3, 1-4	Isolation valves		0.10 *	0.9932	2	0.999999	0.4
1.4, 1-4	Check valves		0.50 *	0.967	2	0.999994	0.4
	Totals			0.932	8	0.99998	8.8

*More than one required; 2 for 1.2.1, and 4 for others

Equipment items 1. 2. 1-4, 1. 3. 1-4 and 1. 4. 1-4 all have an "any-two-of-four" operating relationship. Using Figure 3-8 as described in Section 3. 0, the inherent reliabilities of these items are computed as R 1. 2 = 0. 97, R 1. 3 = 0. 9932 and R 1. 4 = 0. 967. The total group inherent reliability when computed as the product of these four factors — the total group operating relationship indicated by Figure 4. 8 is as follows:

$$R\ 1.0 = 0.999989 \times 0.97 \times 0.999954 \times 0.999940 = 0.9316$$

The flow control valves are obviously the weak links and the item requiring the most attention. The space radiator tube sets are not amenable to sparing or repair provisions. Their inherent reliability of 0. 999989 does exceed the equal risk level requirement of 0. 9999885 or, no more than 11 failures in 10^6 missions.

The second subgroup consists of four radiator flow controls which are interchangeable. These flow control valves work in conjunction with the tube sets that will be used two at a time. All four flow control valves will be maintained in operation, that is, as soon as one fails it will be replaced by a spare so long as spares are available. By taking advantage of the interchangeability characteristic, the flow control function can still continue after all spares have been used and two of the last four valves to be put in operation have failed. This assumes perfect interchangeability and the time required is less than the downtime constraint, up to the last failure. As a result, "any-two-of-four" situation exists for this item. The exposure to failure of four items, each with failure rates of 10. 0 failure/10^6 hours, for a 16,800 hour duty cycle is; a $\lambda t_{1.2} = 4 \times 10.0 \times 10^{-6} \times 16,800 = 0.672$. As sparing capability is provided for this subgroup the reliability will increase as follows, from Figure 3. 8.

Spares Provided		Subgroup Reliability
0	-	0.97
1	-	0.995
2	-	0.99935
3	-	0.999927
4	-	0.9999934

Since a fraction of a spare can not be provided, the initial maintenance plan for this subgroup should call for provisions for a four-spare capability which slightly exceed the 0.9999885 goal.

The remaining two subgroups of the radiator circuit group are "four-of-four" situations on the slightly conservative basis that any failed isolation or check valve in a subgroup will cause it to become non-functional unless the failed valve is replaced or repaired. Again using Figure 3.8 and information from Table 4.1, both subgroup will require provisions for two spares and the reliabilities will be upgraded to 0.9999999 and 0.9999939.

These four subgroups are in series so the provisions for a total of eight spares for the group - no spare radiator tube sets possible, four spare radiator flow control valves, and two each isolation and check valves - will upgrade the group reliability to 0.9999762. The single item goal of 0.9999882 applied to a series of four subgroups results in a 0.9999528 group goal, indicating that the upgraded 0.9999762 will be satisfactory when the total subsystem is evaluated.

2. Refrigerant Circuit. There will be thirty equipment items normally operating in the refrigerant circuit, this total being made up of eleven pairs and two quadruples of the thirteen different kinds of equipment (Table 4.2). This thirty-item group is functionally two fifteen-item groups operating in parallel. The two groups are physically located, one each on the two equipment floors of the mission module. Each are capable of carrying the full load in the event of an equipment failure on the other floor. Since interchangeability exists between like items in the different areas, and in cases of the shutoff and check valves, between like items in the same area, the reliability logic diagram takes the form shown in Figure 4.9.

Table 4.2. Refrigerant Circuit Analysis

Logic Block Number	Component	Duty Cycle (hrs)	Failure Rate x 10^6	Reliability	Number of Spares	Contribution to P_s	Spares Weight (lbs)
2.1, 1-2	Refrigerant pump	16,800	5.0	0.987	2	0.99996	3.0
2.2, 1-2	Pump control		1.0	0.9994	1	0.999994	1.0
2.3, 1-2	Refrigerant evap.		0.17	0.99998	0	0.99998	–
2.4, 1-2	Evap. back press. cont.		1.0	0.9994	1	0.999994	4.8
2.5, 1-2	Evap. back press. valve		0.05	0.999998	0	0.999998	–
2.6, 1-2	Coolant heat exch.		0.17	0.99998	0	0.99998	–
2.7, 1-2	Cabin heat exch.		0.17	0.99998	0	0.99998	–
2.8, 1-2	Cabin temp. cont.		5.0	0.987	2	0.99996	1.8
2.9, 1-2	Evap. water cont.		10.0	0.952	3	0.99997	3.0
2.10, 1-2	Heat exch. diverter		11.0	0.946	3	0.99995	3.0
2.11, 1-2	Ref. temp. cont. valve		5.0	0.987	2	0.99996	1.5
2.12, 1-4	Ref. shutoff valve		0.5	0.967	1	0.999998	0.8
2.13, 1-4	Ref. check valve		0.55	0.963	2	0.999992	0.2
2.19, 1-2	Selector valve		1.1	0.963	2	0.999992	1.2
	Totals			0.776	19	0.99972	20.3

The inherent reliability of the refrigerant circuit group is estimated to be 0.7756, as shown in Table 4.2. This depends on the possibility of cannibalizing systems. The weakest link being the pair of heat exchanger diverters whose combined reliability is only 0.946. However, since more than half of the remaining equipment item subgroups have reliabilities of less than 0.990, the function is obviously not reliable enough in its present form.

A total of nineteen spares, allocated as shown in Table 4.2, will bring the subsystem contribution to P_S to over 0.9997, and, since the equal risk requirement for a group of fourteen is 0.99983, this function will be close enough. Despite the comparatively large number of spares required because of the several weak links (all with approximately the same failure hazard) only about 22 pounds were added to the spares weight.

3. Atmosphere Circuit. The atmosphere circuit is a relatively simple equipment group, made up of twelve applications of three different kinds of items listed in Table 4.3. The cabin blower items 3.1.1-4, will be installed in parallel sets of two — one set on each equipment floor. Normally, one blower on each floor will be in operation. Thus, each blower may be expected to have a duty cycle of about half the total mission hours. On occasions of scheduled, or unscheduled maintenance involving an equipment item failure in this or any of the interdependent groups, neither blower on one floor and both blowers on the other floor will be in operation. Since the blowers are physically identical and interchangeability between floors will exist, an "any-two-of-four" situation is acceptable. Both of the selector valves and all six shutoff valves will be required to operate for the entire mission as represented by the reliability logic diagram of Figure 4.10.

The group reliability was found to be quite low for the mission 0.6149. Again, this low group reliability is due to relatively low and somewhat evenly distributed failure hazard of all the subgroups as shown in Table 4.3. Improvements are required in this system. There are three subgroups in which components are expected to be interchangeable, thus enhancing P_S. The analysis indicated that eleven spares were required to elevate P_s to over 0.99999, which is well above the equal risk level. These components will require a proper packaging concept.

Table 4.3. Atmosphere Circuit Analysis

Logic Block Number	Component	Duty Cycle (hrs)	Failure Rate x 10^6	Reliability	Number of Spares	Contribution to P_s	Spares Weight (lbs)
3.1, 1-4	Cabin blowers	16,800	5.0*	0.987	3	0.999999	2.4
3.2, 1-6	Shutoff valves	16,800	3.4*	0.70	5	0.999998	2.0
3.3, 1-2	Selector valves	16,800	3.4*	0.89	3	0.999994	1.8
	Totals			0.615	11	0.999991	6.2

*More than one used; only two of 3.1 required.

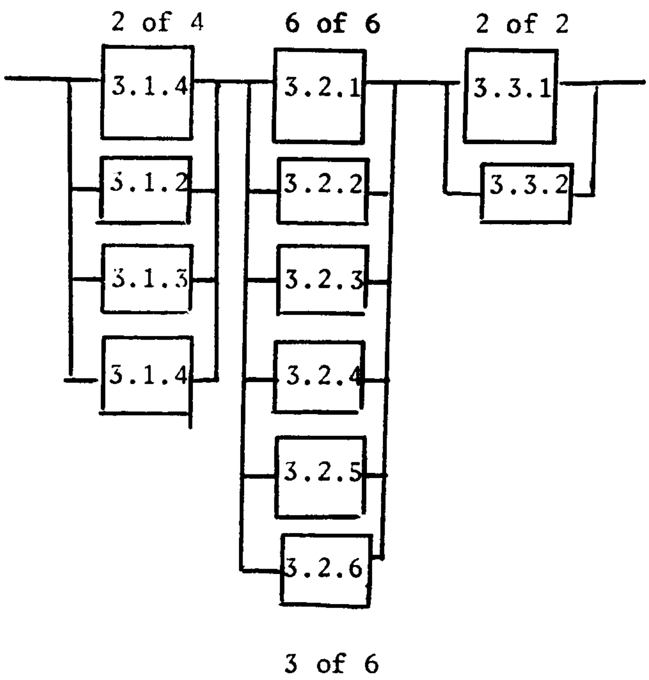


Figure 4.10. Atmospheric Circuit Logic

4. Coolant Circuit. The coolant circuit performs a function for certain kinds of equipment that is very comparable to the function which the atmosphere circuit performs for the crew. This equipment group has seven functional applications of four different kinds of items, as indicated in Figure 4.11 and Table 4.4.

As indicated by Table 4.4, the overall coolant circuit group reliability is 0.8658. The weak link among equipment items is a fluid pump. In this case it is clearly the least reliable item in the group and will require provisions for M and R actions.

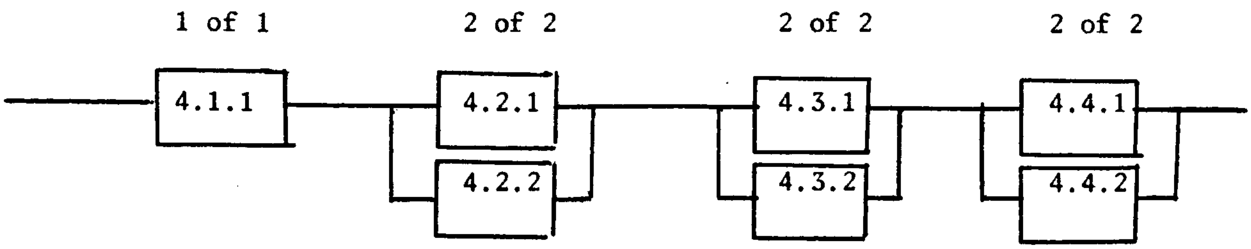


Figure 4.11. Coolant Circuit Logic

Table 4.4. Coolant Circuit Analysis

Logic Block Number	Component	Duty Cycle (hrs)	Failure Rate x 10^6	Reliability	Number of Spares	Contribution to P_s	Spares Weight (lbs)
4.1, 1	Coolant pump	16,800	5.0	0.92	3	0.999998	4.5
4.2, 1-2	Selector valves	16,800	1.1*	0.963	2	0.999992	1.2
4.3, 1-2	Shutoff valves	16,800	0.112*	0.9962	1	0.999993	0.4
4.4, 1-2	Check valves	16,800	0.55*	0.981	2	0.999999	0.2
	Totals			0.866	8	0.99998	6.3

*Two each required

Figure 4. 11 shows the reliability logic diagram for the coolant circuit group which shows that there are no non-interchangeable subgroups. Table 4. 4 indicates that if a total of eight spares is provided and proportioned among the subgroups as listed, this will upgrade the group reliability to 0. 9999820. The group adds only six pounds to the spares complement.

5. Humidity Circuit. The humidity circuit is an eighteen-item group that forms nine interchangeable, "one-of-two," two-item subgroups, as indicated in Figure 4. 12 and Table 4. 5. The inherent reliability of this group is 0. 9243, and this must also be upgraded. In addition, such components as the wicks will have to be replaced periodically to prevent clogging and other resultant damage. The eighteen items form nine interchangeable subgroups of two items each, "one-of-two" situations, each requiring one to three spares to upgrade the group reliability to 0. 9998871. This group reliability of 0. 9998871 is very slightly less than the group equal risk level of 0. 9998930, but is sufficiently close at this point in the analysis.

   This group adds only 31 pounds of spares to the ECLSS Complement. Most of that is created by the catalytic burner, the weight of which is expected to be less in a production system. All components requiring replacement can be exchanged with comparative ease.

6. Water Reclamation. The twenty-item water reclamation equipment group is very similar to the humidity group, in many respects. However, such factors as duty cycle and downtime constraints present a more attractive picture. The system can be down for long periods of time. To maintain conservation, the duty cycles were assumed to be full time, and the reliability was assessed for the function as presented in Table 4. 6, from the reliability logic of Figure 4. 13; component interchangeable is as indicated. The resultant function reliability was also too low at 0. 924. The two weakest links are the catalytic burner control and the pump.

   The water reclamation group is the first in which the maintenance logic diagram differs significantly from the standard reliability logic diagram by virtue of a non-interchangeable subgroup. Items 6. 1. 1, 6. 2. 1, 6. 3. 1 and 6. 4. 1 form an interdependent subgroup on equipment floor A while items 6. 1. 2, 6. 2. 2, 6. 3. 2 and 6. 4. 2 form a similar subgroup on equipment floor B. Physical limitations dictate then that within this subgroup, items cannot be interchanged between floors. If these four items are

Table 4.5. Humidity Circuit Analysis

Logic Block Number	Component	Duty Cycle (hrs)	Failure Rate x 10^6	Reliability	Number of Spares	Contribution to P_s	Spares Weight (lbs)
5.1, 1-2	Humidity control	16,800	2.5*	0.9967	2	0.999998	0.5
5.2, 1-2	Catalytic burner	16,800	1.7*	0.9984	1	0.99997	10.4
5.3, 1-2	Cat. burner control	16,800	10.0*	0.952	3	0.98997	4.0
5.4, 1-2	Filter	16,800	0.5*	0.99986	1	0.9999993	0.3
5.5, 1-2	Fan	16,800	3.0*	0.9952	2	0.999996	2.6
5.6, 1-2	Charcoal filter	16,800	1.0*	0.99944	1	0.999994	1.0
5.7, 1-2	Pump	16,800	5.0*	0.987	2	0.99996	4.0
5.8, 1-2	Waste receiver	16,800	0.22*	0.9997	1	0.999999	0.6
5.9, 1-2	Selector valve	16,800	3.4	0.994	2	0.999994	2.8
	Total				15	0.99989	31.2

*Two each used; only one required

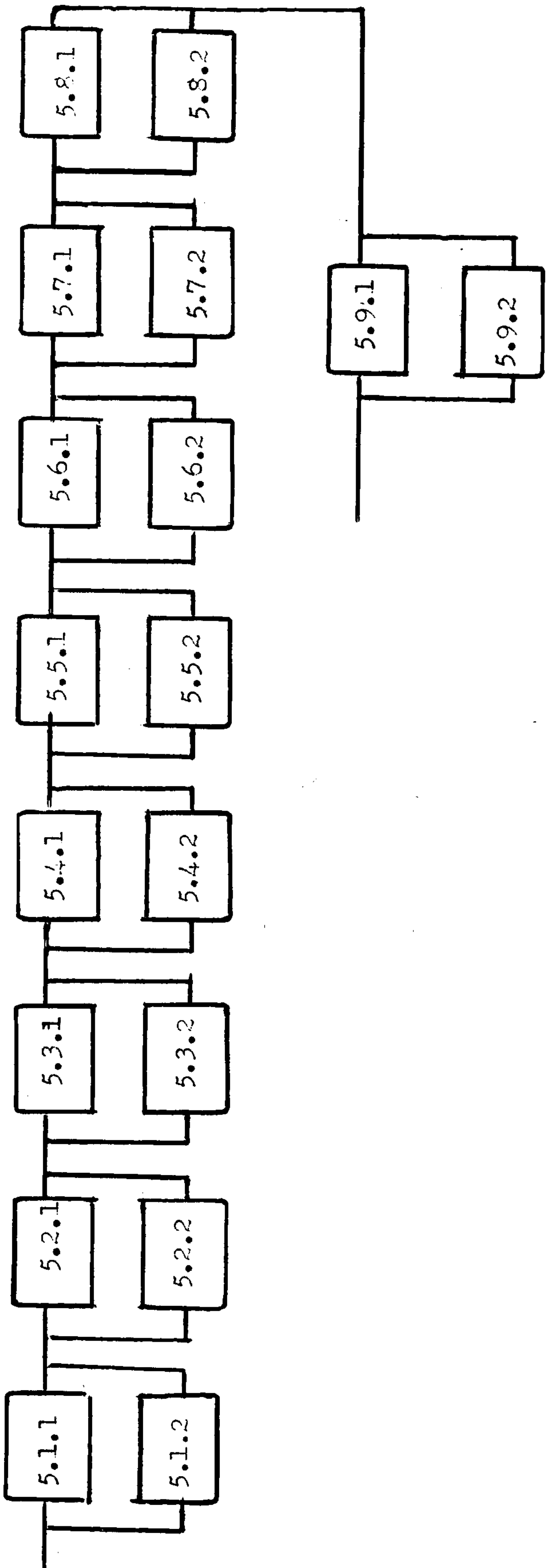


Figure 4.12, Humidity Circut Reliability Logic

Table 4.6. Water Reclamation Loop Analysis

Logic Block Number	Component	Duty Cycle (hrs)	Failure Rate x 10^6	Reliability	Number of Spares	Contribution to P_s	Spares Weight (lbs)
6.1, 1-2	Condenser	16,800	0.17		1	0.99998	5.2
6.2, 1-2	Recuperator	16,800	0.17		1	0.99998	5.2
6.3, 1-2	Pyro reactor	16,800	10.0*	0.974	3	0.99997	5.2
6.4, 1-2	Evaporator	16,800	0.17		1	0.99998	5.2
6.5, 1-2	Pressure regulator	16,800	1.0*	0.9994	1	0.999994	1.0
6.6, 1-2	Heater control	16,800	7.67*	0.973	3	0.999992	2.7
6.7, 1-2	O_2 Flow control	16,800	0.10*	0.99999	0	0.999995	-
6.8, 1-2	Check value	16,800	0.55	0.946	3	0.999999	0.3
	Totals			0.838	13	0.9999	24.8

*Two each used of all components; only one of these required.

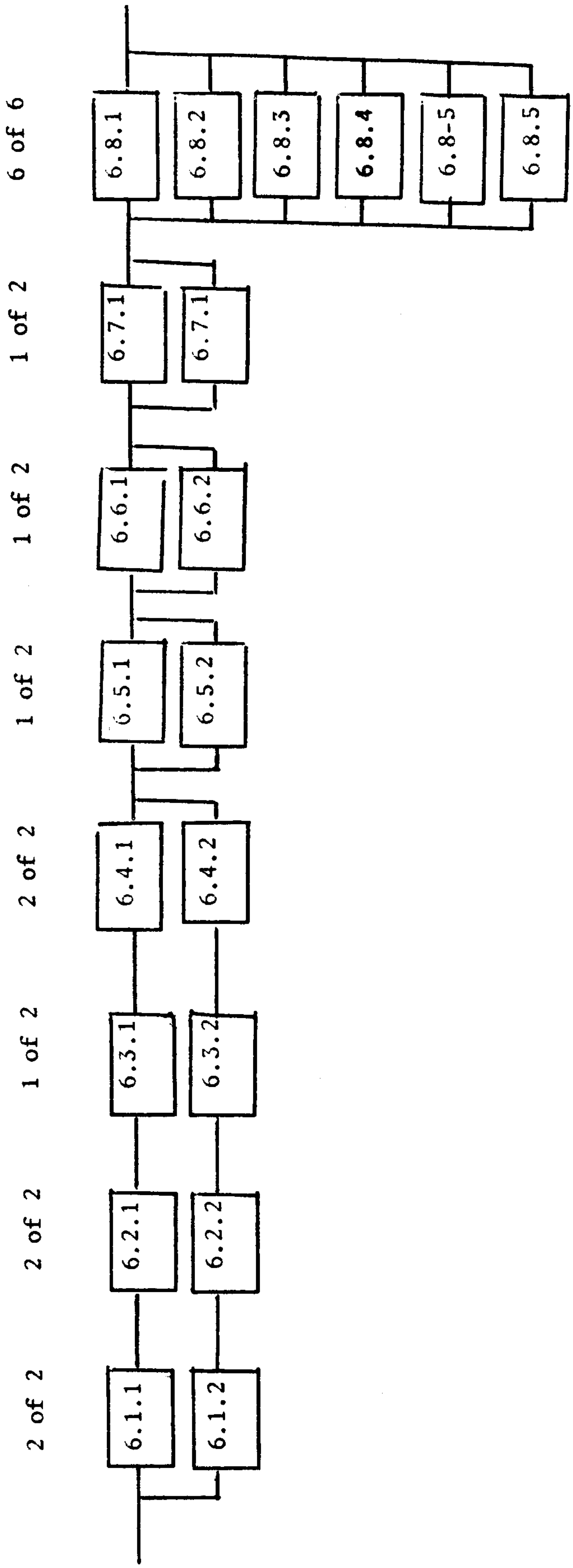


Figure 4.13. Water Reclamation Loop Logic

considered separately, as in Figure 4.21, a beginning upon a maintenance logic diagram can be made. For crew safety and mission success, at least one of these subgroups must be operative; conversely, one may be allowed to fail. This condition is met if no more than one of the eight items is allowed to run out of spares. It follows that one of the four kinds of items may be a "one-of-two" situation, but the other three must be "two-of-two" situations.

Inspection of columns three through eight of Table 4.6 reveals that of these four items, 6.3.1 and 6.3.2 clearly presents the greatest exposure to failure. Therefore it should be the one assigned the "one-of-two" classification. The remaining four kinds of items in the group form three interchangeable subgroups of two items, and one of six items, and will appear in the group maintenance logic diagram in Figure 4.13. Based upon this diagram, group reliability of 0.9999056 can be obtained by providing thirteen spares, as listed in column eleven of Table 4.6. Note that this is slightly above the group equal risk level of 0.9999882 = 0.9999001. This added 24 pounds to the spares complement.

7. Potable Water Storage

The potable water storage equipment group includes items not on either equipment floor. In this preliminary design concept, the two potable water storage tanks are not expected to be accessible for repair, or require replacement. There will be two tanks, either of which could provide the minimum storage requirements necessary to achieve the 0.99 P_s for the mission. There are no unusual limitations or characteristics associated with the remaining items in this group. The reliability logic and associated data is presented in Figure 4.14 and Table 4.7, respectively. The reliability without repair will exceed 0.98. However, improvements are desirable. The water tanks are inaccessible and are not amenable to sparing. Providing one spare each to the other three kinds of items will improve the group reliability to an acceptable 0.9999958 - 0.9999958 vs $(0.9999916)^4 = 0.9999664$. The added weight is just over one pound.

8. Waste Water Storage

The waste water storage group is very similar to the potable water storage group since the water tanks are not expected to be accessible for repair or replacement. It differs from the potable water storage group in that no water outlets exist for crew use, or

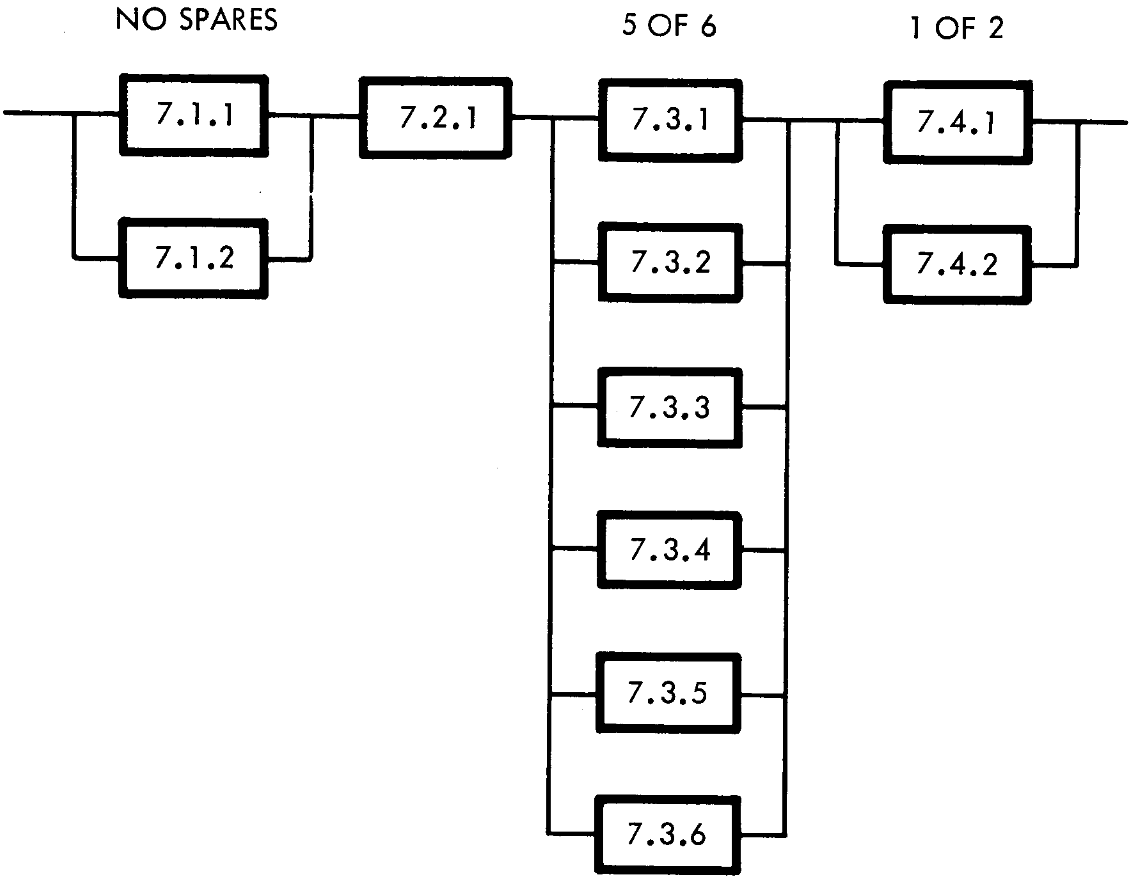


Figure 4.14. Potable Water Storage Logic

Table 4.7. Potable Water Storage Analysis

Logic Block Number	Component	Duty Cycle (hrs)	Failure Rate x 10^6	Reliability	Number of Spares	Contribution to P_s	Spares Weight (lbs)
7.1, 1-2	Water tank	16,800	0.10*	0.999998	0	0.999998	-
7.2, 1	Check valve	16,800	0.10	0.98410	1	0.999998	0.1
7.3, 1-6	Shutoff valve	16,800	0.10*	0.99995	1	0.9999998	0.4
7.4, 1-2	Potable water feed	16,800	0.50*	0.99987	1	0.9999993	0.6
	Totals			0.984	3	0.999996	1.1

*More than one used; only one of these required.

associated valves. The reliability logic and associated information for this group is presented in Figure 4.15 and Table 4.8. It also requires some improvement since the overall reliability is a little better than 0.977.

This group represents the minimum in maintenance plan complexity. As in the case of the last group, water storage tanks are not amenable to maintenance. One spare check valve and two spare shutoff valves, as listed in Table 4.8, will upgrade the reliability of this group to 0.9999966. Less than one pound is added to the spares list.

9. Carbonization Cell. The carbonization cell equipment group is the largest in terms of number of items — there are a total of seventy-one applications of nineteen different kinds of items. Four interchangeable subgroups of two, one of six, one of sixteen, one of eighteen, and a subgroup of one (a single item serving both floors) operate in series with the two non-interchangeable subgroups. Figure 4.16 and Table 4.9 show the reliability logic diagram and associated data. These relationships and the equipment item information for this group. This function reliability is only rated at 0.86 and must operate full time with limited time down for M & R.

While the carbonation cell group has a greater variety of equipment items than most other groups, it does not present the greatest spares requirement, either in terms of number of spares or total weight of spares. Table 4.9 shows that providing twelve spares will result in a group reliability of 0.999731 which is sufficiently close to the group equal risk level of 0.9997994. Almost 28 pounds of spares were added to the list, most of which is attributable to the CO_2 pump.

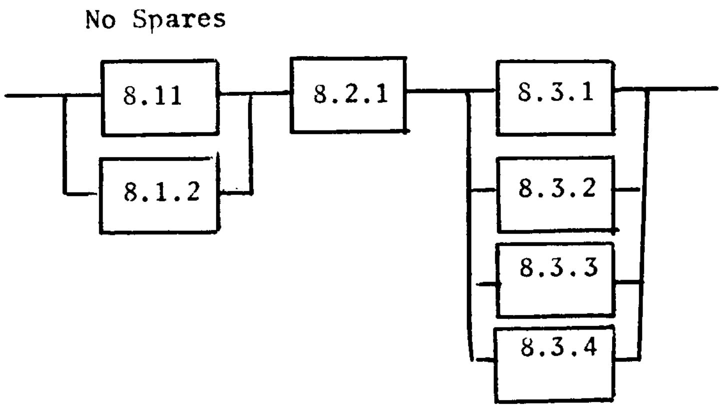


Figure 4.15. Waste Water Logic

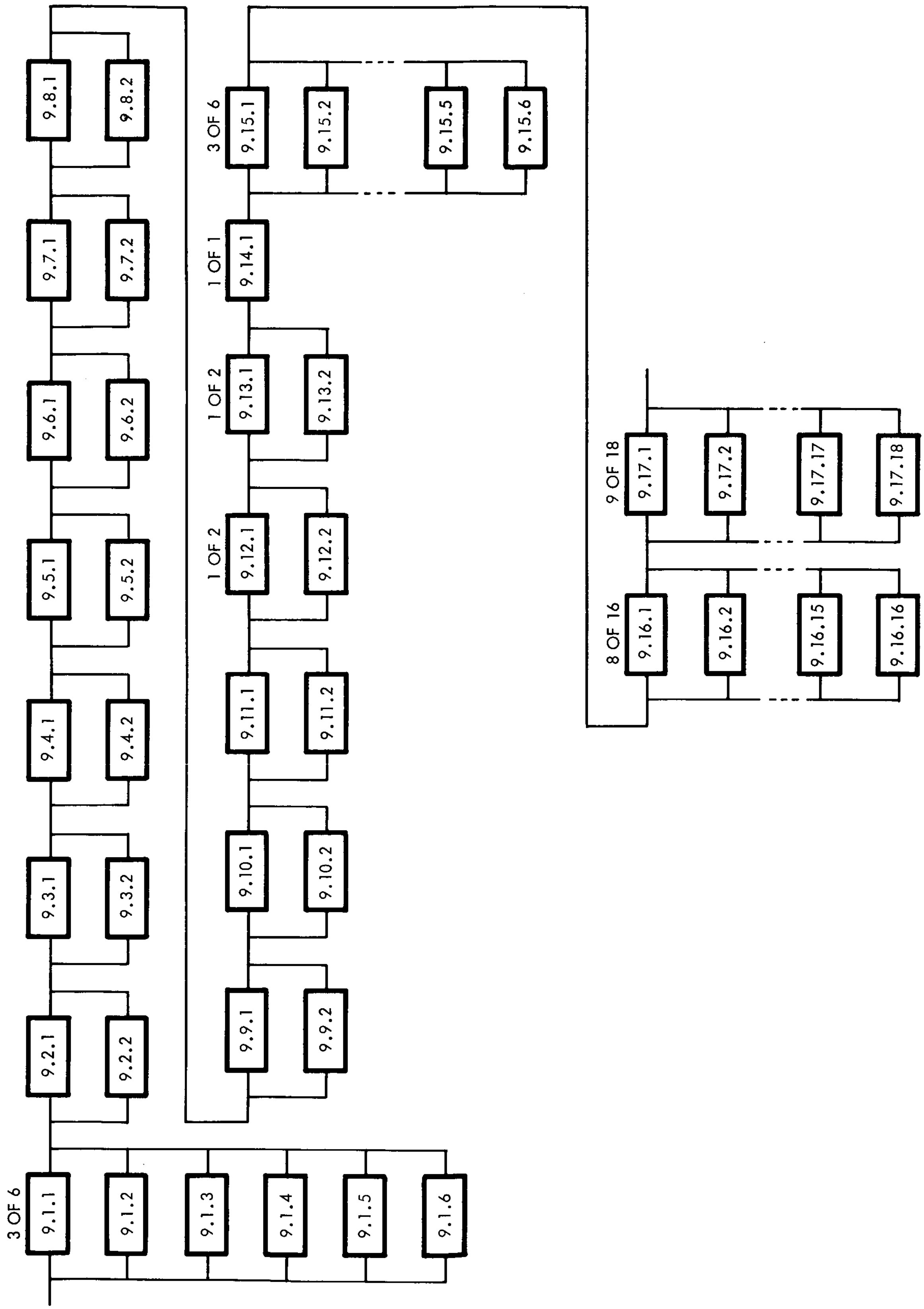


Figure 4.16. Carbonization Cell Logic

Table 4.8. Waste Water Storage Analysis

Logic Block Number	Component	Duty Cycle (hrs)	Failure Rate x 10^6	Reliability	Number of Spares	Contribution to P_s	Spares Weight (lbs)
8.1, 1-2	Water tank	16,800	0.1*	0.999998	0	0.999998	-
8.2.1	Check value	16,800	0.1*	0.984	1	0.999998	0.1
8.3, 1-4	Shutoff valve	16,800	0.1*	0.993	2	0.999999	0.8
	Totals			0.977	3	0.9999966	0.9

*More than one used; only one of these required.

Table 4.9. Carbonization Cell Analysis

Logic Block Number	Component	Duty Cycle (hrs)	Failure Rate x 10^6	Reliability	Number of Spares	Contribution to P_s	Spares Weight (lbs)
9.1, 1-6	Stage	16,800	0.25*	0.999999	0		–
9.2, 1-2	CO_2 condenser	16,800	0.17	0.99998	0		–
9.3, 1-2	Condenser	16,800	0.17	0.99998	0		–
9.4, 1-2	Cationic exch.	16,800	0.25	0.99996	0		–
9.5, 1-2	Anionic exch.	16,800	0.25	0.99996	0		–
9.6, 1-2	Carcoal bed	16,800	0.10	0.999995	0		–
9.7, 1-2	Filter	16,800	0.50	0.99986	1	0.9999993	0.3
9.8, 1-2	Water tank	16,800	0.10	0.999995	0		–
9.9, 1-2	CO_2 tank	16,800	0.044	0.999999	0		–
9.10, 1-2	Fan	16,800	5.0	0.987	2	0.99996	2.6
9.11, 1-2	Water pump	16,800	5.0	0.987	2	0.99996	4.0
9.12, 1-2	CO_2 pump	16,800	9.0	0.963	3	0.99998	15.0
9.13, 1-2	Relief valve	16,800	2.0	0.936	2	0.99995	4.8
9.14, 1-	Selector valve	16,800	1.1	0.981	2	0.999999	1.2
9.15, 1-6	Water meter dev.	16,800	1.0*	0.999996	0		–
9.16, 1-16	Shutoff valve	16,800	0.1*	0.999999	0		–
9.17, 1-18	Check valve	16,800	0.1*	0.999999	0		–
	Totals			0.861	12	0.99973	27.9

*More than one required; see logic diagram.

10. Cryogenic Oxygen Supply. Figure 4.17 and Table 4.10 present the reliability logic information for the cryogenic oxygen supply equipment group. There are no non-interchangeable subgroups, eight interchangeable subgroups of two, one of four, one of six and one subgroup of eight. A partial backup to the function of this group is provided by groups 12.0 and 13.0 where O_2 recovery is possible and will relieve the emergency. The basic reliability is expected to be very low, 0.459, because of the long list of weak links.

The cryogenic oxygen supply group has only one equipment item limitation which might require special consideration in the development of a maintenance plan, the inaccessibility of the cryogenic storage tanks. The tanks proper have so low a failure rate — 0.000001 failures/10^6 hours – as to have effectively no exposure to failure. However, the pressure transducers, which are often physically located at the tanks, have the highest failure rate of the group – 23.2 failures/10^6 hours. This is an example of a situation in which "provision of spares" must be taken in the broad sense, and provisions made for sparing action. Also location in an accessible area must be considered and the necessary isolation valves, etc., must be designed into the subsystem as well as actually providing the appropriate number of spares for replacement of failed items. Table 4.10 indicates that providing a total of twenty-three spares will upgrade the group reliability to 0.99996836 which is comfortably above the 0.9999882 = 0.9998702 equal risk level, 22 pounds of spares were added.

11. High Pressure Oxygen Supply. The high-pressure oxygen supply equipment group has thirty-one applications of eight different kinds of equipment items. All are in interchangeable subgroups, one of fourteen items, one of six, five of two, and one group of one or a single item serving both floors. The reliability logic diagram and equipment item information are presented in Figures 4.18 and Table 4.11. The reliability os 0.856 is considered inadequate and M and R is recommended.

    Some similarity exists between the high pressure and cryogenic oxygen supply groups since both have two inaccessible storage tanks, but a basic difference also exists in terms of the criticality of these tanks. It is assumed that although considerable inconvenience might be involved, one of the two high pressure oxygen storage tanks would provide for crew safe return with reduced activity. Table 4.11 indicates that provision of thirteen spares, as listed in column eleven, will upgrade the group reliability to 0.9999709 which is acceptable when compared to the goal 0.9999882, the equal risk level for the group.

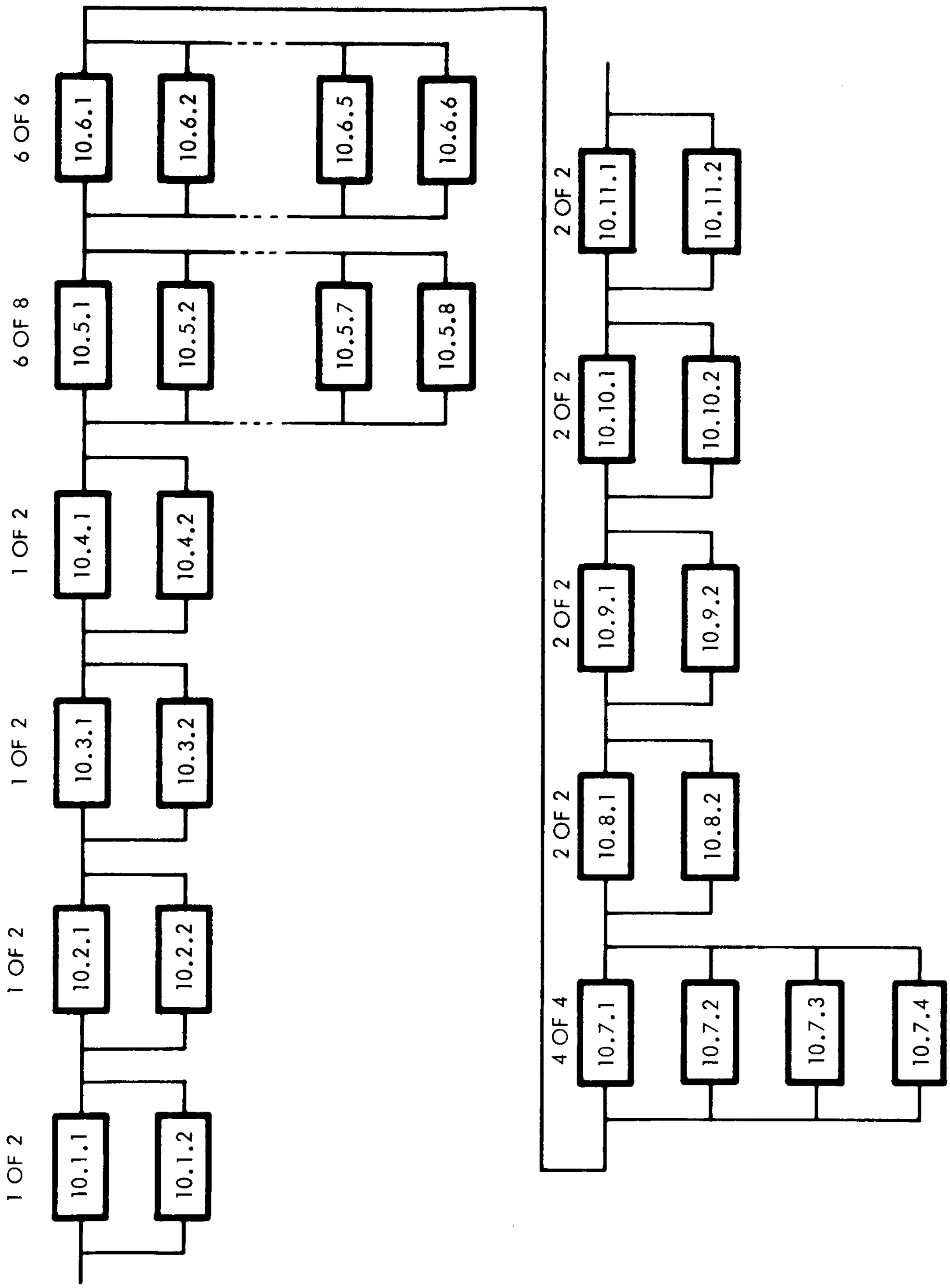


Figure 4.17. Cryogenic Oxygen Supply Logic

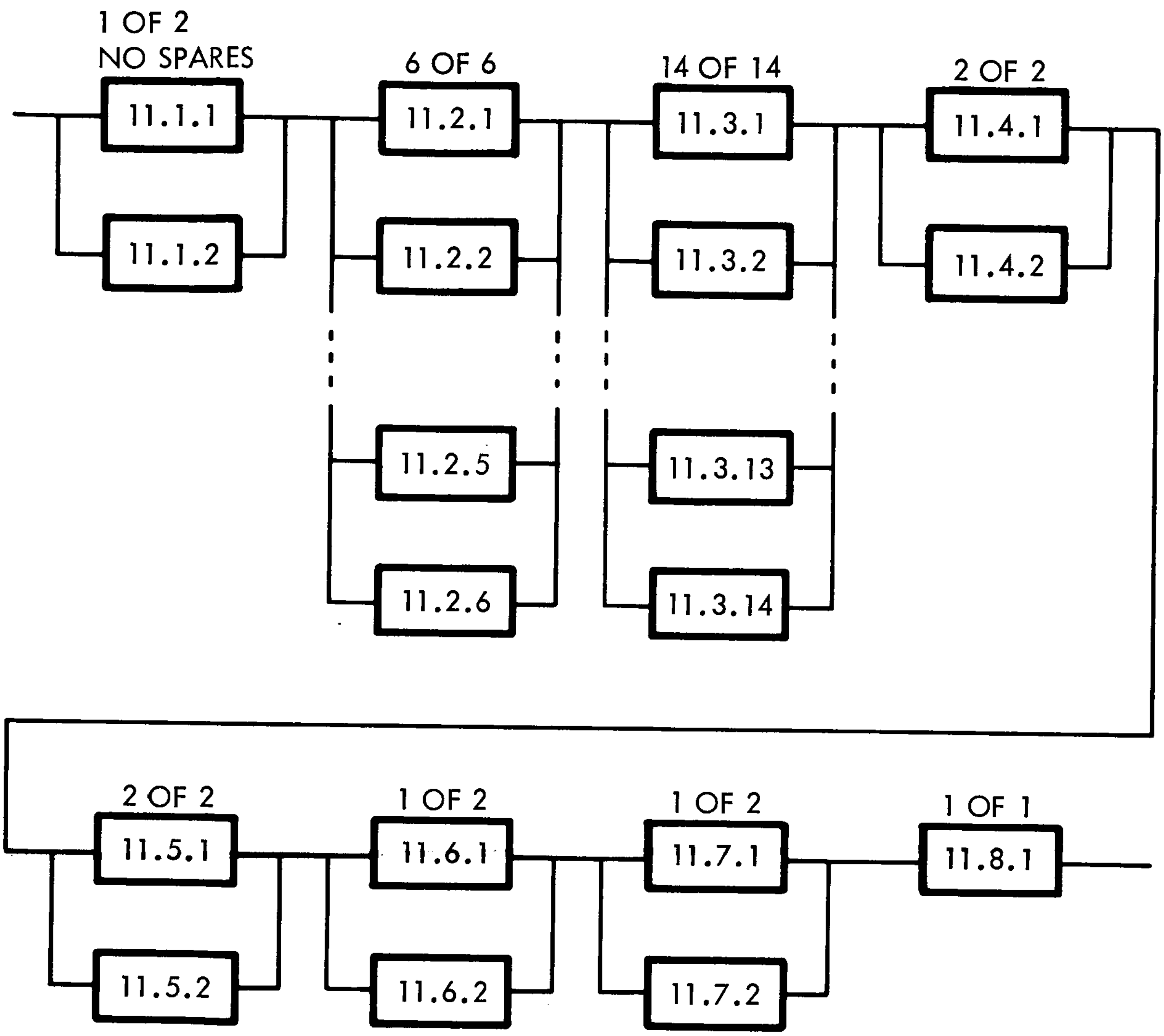


Figure 4.18. High-Pressure Oxygen Supply Logic

Table 4.10. Cryogenic Oxygen Supply Analysis

Logic Block Number	Component	Duty Cycle (hrs)	Failure Rate x 10^6	Reliability	Number of Spares	Contribution to P_s	Spares Weight (lbs)
10.1, 1-2	O_2 Part. press. cont.	16,800	1.0*	0.9994	1	0.999994	0.8
10.2, 1-2	O_2 partial press. sensor	16,800	1.0*	0.9994	1	0.999994	0.2
10.3, 1-2	Display	16,800	0.6*	0.9998	1	0.999999	2.0
10.4, 1-2	Regulator	16,800	1.0*	0.9994	1	0.999994	1.0
10.5, 1-2	Check valve	16,800	0.1	0.987	2	0.9999996	0.2
10.6, 1-2	Shutoff valve	16,800	0.1	0.990	2	0.999999	0.8
10.7, 1-2	Selector valve	16,800	1.1	0.928	3	0.999999	4.2
10.8, 1-2	Heater control	16,800	7.67	0.773	4	0.999992	3.6
10.9, 1-2	Relief valve	16,800	2.0	0.936	3	0.9999992	7.2
10.10, 1-2	Press. transducer	16,800	10.0	0.70	5	0.999998	2.5
10.11, 1-2	Cryo. O_2 tank	16,800	1×10^{-12}*	1.0	0		-
	Totals			0.458	23	0.999968	22.5

*More than one of all components used; only one of these are required.

Table 4.11. High Pressure Supply Analysis

Logic Block Number	Component	Duty Cycle (hrs)	Failure Rate x 10^6	Reliability	Number of Spares	Contribution to P_s	Spares Weight (lbs)
11.1, 1-2	High press. O_2 task	16,800	0.044*	0.999999	0	0.999999	-
11.2, 1-6	Check valve	16,800	0.1	0.990	2	0.999999	0.2
11.3, 1-4	Shutoff valve	16,800	0.1	0.976	2	0.999998	0.8
11.4, 1-2	Selector valve	16,800	1.1	0.963	2	0.999992	2.8
11.5, 1-2	Relief valve	16,800	2.0	0.936	3	0.9999992	7.2
11.6, 1-2	Regulator	16,800	1.0*	0.9994	1	0.999994	1.0
11.7, 1-2	Regulator	16,800	1.0*	0.9994	1	0.999994	1.0
11.8, 1	Regulator	16,800	1.0	0.984	2	0.9999993	2.0
	Totals			0.856	13	0.99997	15.0

*Two or more of these components are used; only one of these required.

12. Bosch Reactor. In terms of EC/LSS functions, the Bosch reactor equipment group is only half of the oxygen regeneration functional groups because it must operate in conjunction with an electrolysis cell functional group. However, for ease of analysis, the two are considered as separate groups. There exist fifty-one applications of seventeen different kinds of items, no non-interchangeable subgroups, ten subgroups of two interchangeable items, six subgroups of four interchangeable items, and one subgroup of seven interchangeable items. The reliability logic diagram and the equipment item information are presented in Figure 4.19 and Table 4.12. Since the system operates only as a backup to the crogenic storage, the O_2 regenerator was assumed to require about a 10-percent duty cycle, or 1680 hours. Since P_s is to be calculated without the capabilities of this system, a lower reliability could be tolerated; but, 0.92 was considered too low and spares are recommended. Therefore, on the basis of the assumed duty cycle, the ten spares weighing almost 24 pounds were required to achieve $P_s = 0.99952$. With a reasonable design, all of the components requiring a maintenance action are expected to be within the work envelope of the crew.

13. Electrolysis Cell. The final functional equipment group into which the EC/LSS has been divided is the electrolysis cell group. There are 117 applications of 17 different kinds of equipment items. No subgroups of non-interchangeable items are formed, one subgroup of 40 interchangeable items, one subgroup of 36 interchangeable items, two subgroups of 8, twelve subgroups of 2, and one subgroup of one are formed. The reliability logic diagram and equipment information are presented in Figure 4.20 and Table 4.13. This function is also part of the O_2 recovery system not included in P_s. The reliability of 0.981 is not considered adequate without some form of M and R. Based upon the same arbitrary 1680-hour duty cycle as the Bosch Reactor with which it works, the total spares required is 8, proportioned as listed in Table 4.13. When provided with these eight spares, the group is upgraded to about 0.9998, a figure well above the 0.9990 requirement for a complete subsystem. This also adds about 25 pounds to the spares complement. Thus the design can be configured to accommodate easily the M and R requirements within any time constraints.

4.1.4 Alternate Operational Modes for the ECLSS

Because of the criticality of the ECLSS, it is necessary to provide the crew with every assurance of safe return; in particular, provide adequate backup modes for the ECLSS to satisfy all reasonable contingencies. Such is

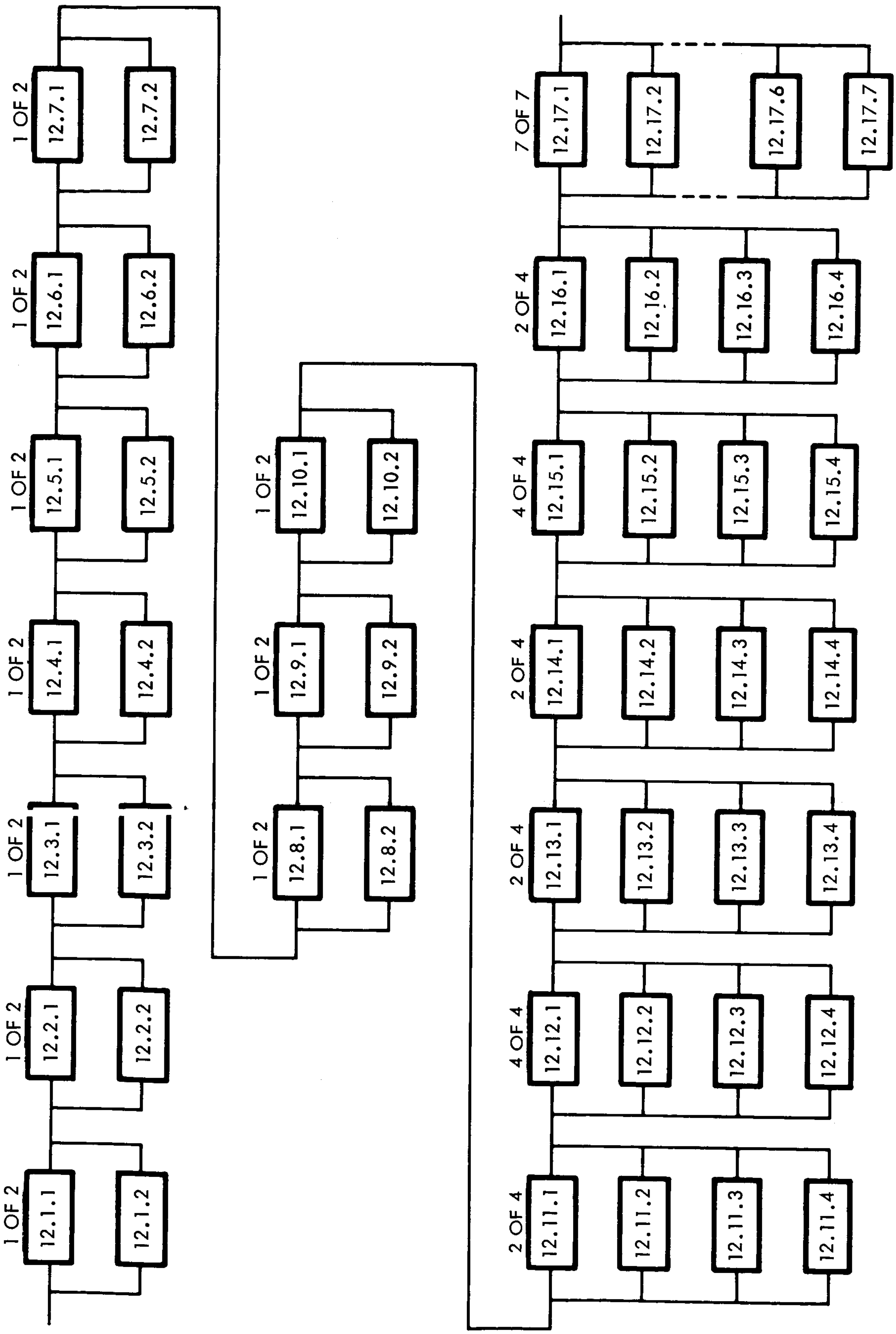


Figure 4.19. Bosch Reactor Logic

Table 4.12. Bosch Reactor Analysis

Logic Block Number	Component	Duty Cycle (hrs)	Failure Rate x 10^6	Reliability	Number of Spares	Contribution to P_s	Spares Weight (lbs)
12.1, 1-2	Bosch reactor	1,680	20.0	0.9978	1	0.999954	5.0
12.2, 1-2	Drive motor	1,680	4.5	0.9999	0		-
12.3, 1-2	Heater control	1,680	7.67	0.9997	1	0.999997	0.9
12.4, 1-2	Blower	1,680	3.0	0.9953	1	0.9998	3.5
12.5, 1-2	Expendable filter	1,680	0.55	0.9978	1	0.99995	3.5
12.6, 1-2	Blower	1,680	20.0	0.999999	0		-
12.7, 1-2	Recuperator	1,680	0.17	0.999999	0		-
12.8, 1-2	Condenser	1,680	0.17	0.999999	0	0.999999	5.0
12.9, 1-2	H_2 pump	1,680	9.0	0.99955	1	0.999999	5.0
12.10, 1-2	H_2 tank	1,680	0.044	0.999999	0		-
12.11, 1-4	Flow control	1,680	0.1	0.999999	0		-
12.12, 1-4	Relief valve	1,680	2.0	0.987		0.99991	2.4
12.13, 1-4	Selector valve	1,680	1.1	0.999999	0		-
12.14, 1-4	Carbon filter	1,680	0.55	0.999999	0		-
12.15, 1-4	Shutoff valve	1,680	3.4	0.977	2	0.999998	0.8
12.16, 1-4	Check valve	1,680	0.1	0.999999	0		-
12.17, 1-7	Selector valve	1,680	3.4	0.961	2	0.99999	2.8
	Total			0.92	10	.99952	23.9

Table 4.13. Electrolysis Cell Analysis

Logic Block Number	Component	Duty Cycle (hrs)	Failure Rate x 10^6	Reliability	Number of Spares	Contribution to P_s	Spares Weight (lbs)
13.1, 1-2	Charcoal bed	1,680	0.1	0.9999999	0		
13.2, 1-2	Anionic bed	1,680	15.0	0.9987	1	0.99998	8.0
13.3, 1-2	Cationic bed	1,680	15.0	0.9987	1	0.99998	8.0
13.4, 1-2	Water tank	1,680	0.1	0.9999999	0		
13.5, 1-2	Water pump	1,680	5.0	0.99986	0		
13.6, 1-2	H_2 Steam cond.	1,680	0.17	0.9999999	0		
13.7, 1-2	O_2 Steam cond.	1,680	0.17	0.9999999	0		
13.8, 1-2	O_2 Press. reg.	1,680	1.0	0.9999999	0		
13.9, 1-2	Mod. pr. cont.	1,680	2.9	0.9999994	0		
13.10, 1-2	Relief valve	1,680	2.0	0.9932	1	0.999972	2.4
13.11, 1-2	Cell press. reg.	1,680	11.0	0.99935	1	0.999992	2.4
13.12, 1-2	Flow control	1,680	10.0	0.99944	1	0.999994	2.0
13.13, 1-8	Flow control	1,680	0.1	0.999999	0		
13.14, 1-8	Module	1,680	15.0	0.9999975	0		
13.15, 1-40	Shutoff valve	1,680	0.1	0.9993	1	0.9999999	0.4
13.16, 1-36	Check valve	1,680	0.1	0.994	1	0.99998	0.1
13.17, 1	Selector valve	1,680	1.1	0.9982	1	0.999998	1.4
	Totals	—	—	0.981	8	0.9998	24.7

the case with the recommended system concept. There are three operational modes possible, primary, backup and emergency (Reference 4.3). These are presented in tabular form in Table 4.14 in logic form in Figures 4.20 through 4.22; a description of each follows:

Table 4.14. Selected EC/LSS Operating Configurations

Subsystem	Operating Configuration		
	Primary	Backup	Emergency
Thermal control	Coolant loop and radiators. One pump operating	Redundant loop plus spares. One pump operating	Water Boiling. One pump operating
Humidity control and air temperature control	Both air circulation loops operating. One blower in each redundant unit running	One redundant unit only operating; both circulation blowers operating; one humidity control blower only running	Allow environmental maximum for short periods of time
CO_2 Removal Molecular sieve or carbonation cell.	Both CO_2 removal units operating at less than maximum capacity	One redundant unit operating at maximum capacity	One redundant unit operating
Contaminant burner	Two units operate at reduced capacity	One unit operating	Cabin venting and monitoring
Water management, (vacuum distillation/pyrolysis) recovery	Two reclamation units operating at reduced capacity	One Reclamation unit operating at maximum capacity	Use stored water at reduced rate
Atmospheric supply	Cryogenic storage for metabolic and leakage. High-pressure storage for repressurization	Two O_2 recovery units operating at less than maximum capacity (Bosch system)	One O_2 recovery unit operating or high pressure storage

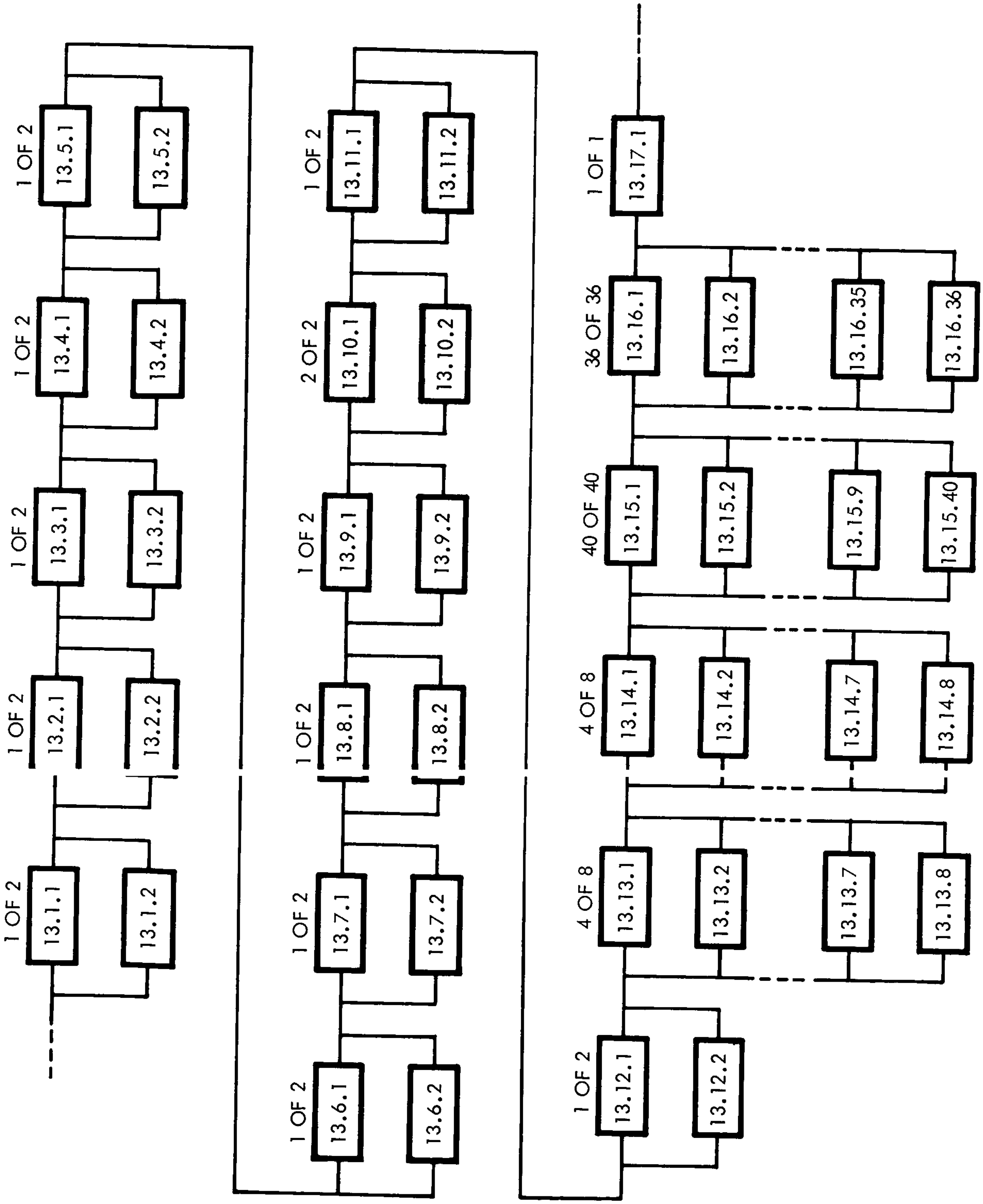


Figure 4.20. Electrolysis Cell Logic

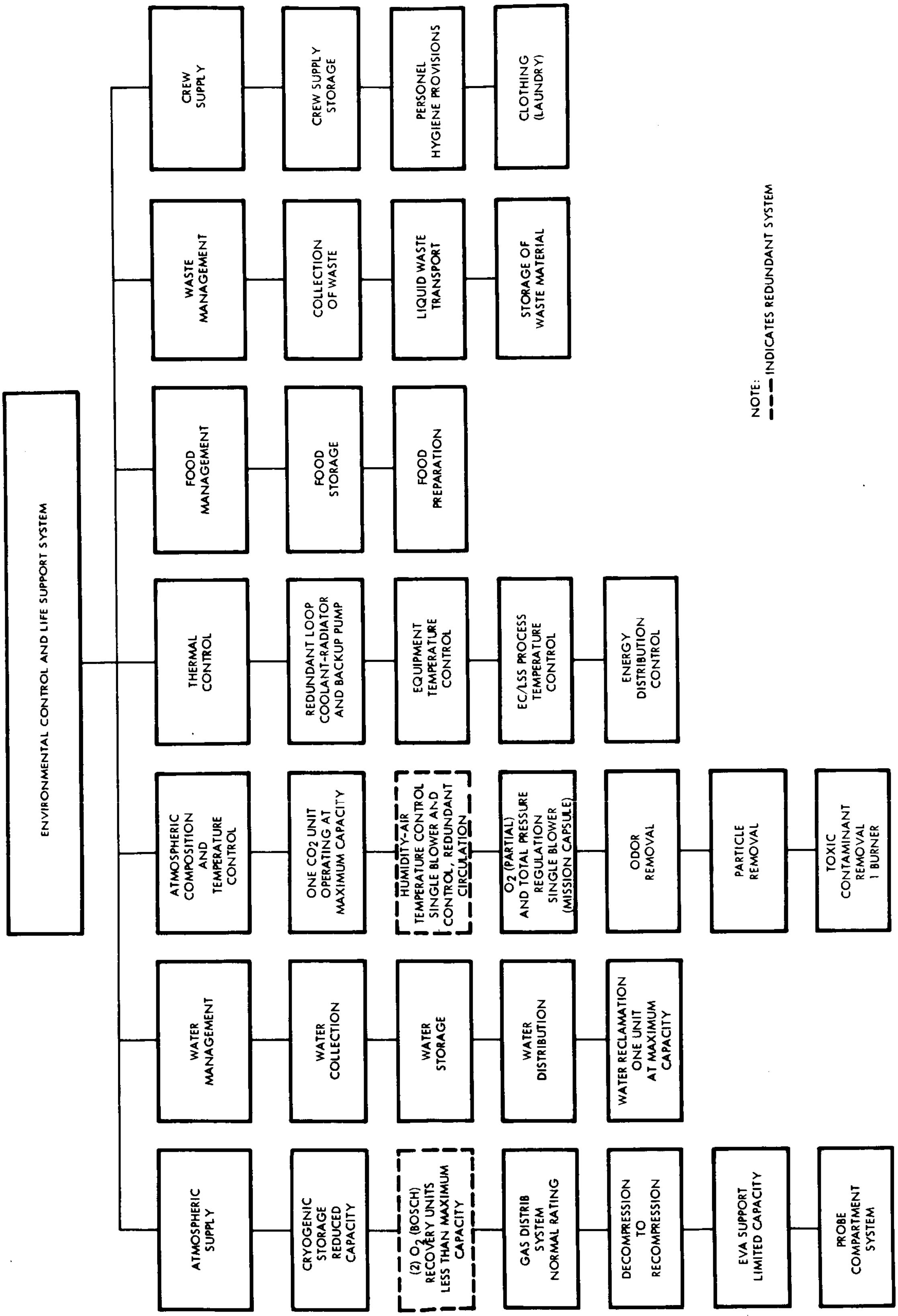


Figure 4.21. Mode B-Backup Operating Mode

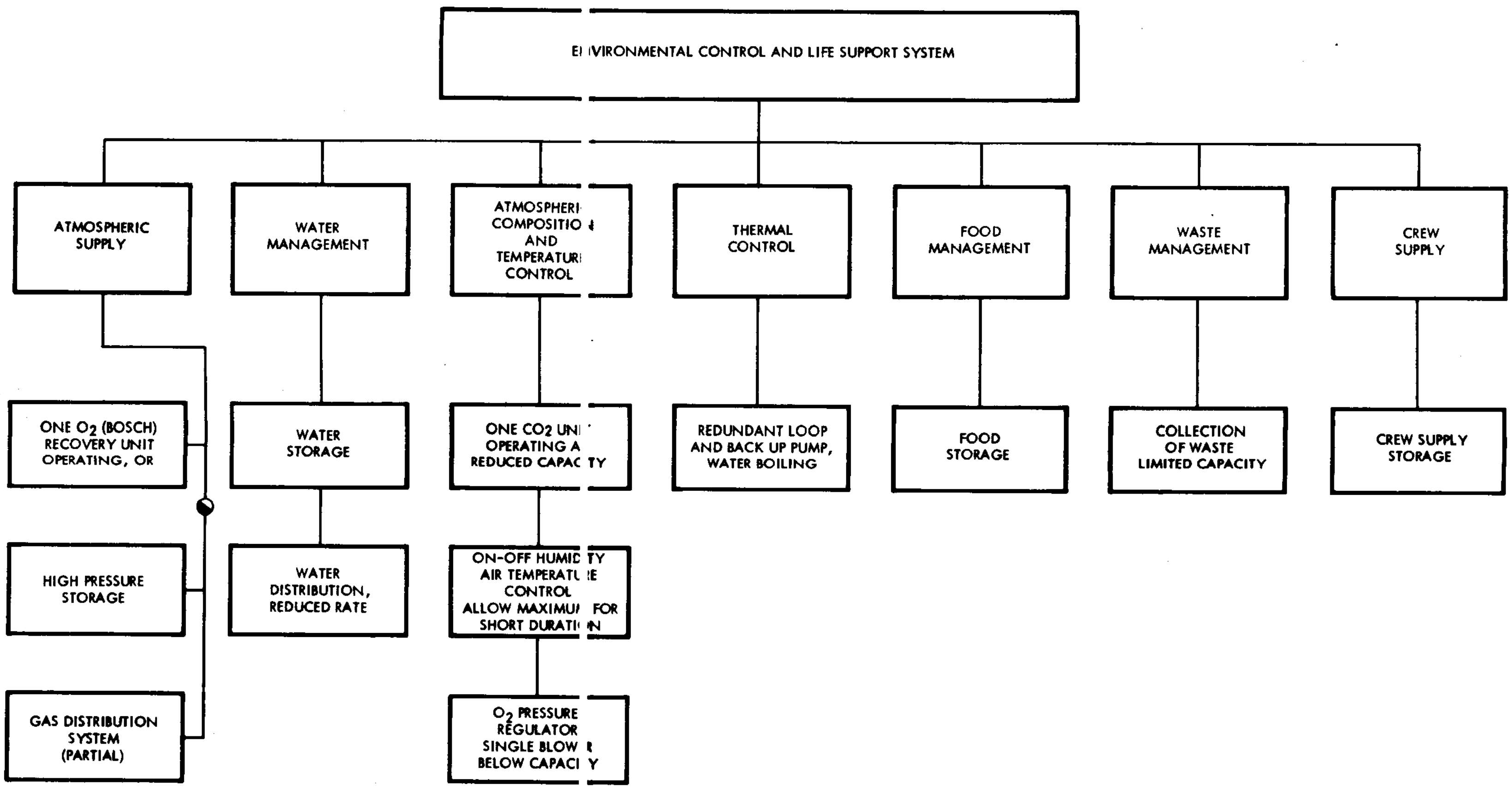


Figure 4.22. Mode C-Emergency Operating Mode

Mode A — Primary Operation Mode

The system configuration for the primary operation mode is shown in Figure 4.20. The function of the atmospheric supply system is to provide two types of gas flow. One flow demands a relatively constant, low-quantity rate for the total mission. The second one is the high-volume flow of gases for short periods of time to meet repressurization requirements. The subcritical cryogenic storage concept meets the constant-flow situation with minimum weight and volume requirements. To satisfy the high-volume gas flow demand for repressurization, a high-pressure gas storage is also required. Hence, both means of storage are shown in the chart. As a backup source of metobolic oxygen, in the event of a primary system shutdown (Mode B), oxygen recovery by the Bosch process is recommended for Mars flyby missions.

The water management system is designed to collect, store, distribute, and reclaim (though not from feces) H_2O for crew support and mission module operation. Redundant water recovery circuit design with parallel water distillation units equipped with vapor pyrolysis, to minimize the contamination and bacterial growth, has been recommended.

The atmospheric composition and temperature control system is based, at this time, on the selection of N_2 over He as a second constituent of atmospheric gas, although the He approach would have saved some weight. The O_2 and total pressure regulation subsystem is based on the priority scheme (Reference 1-2). Humidity — air temperature control functions are accomplished by two identical loops, with both loops operating in the primary mode. Also, two carbon dioxide removal units (carbonation cells) are provided, with both cells operating in the primary mode below maximum capacity.

The thermal control is designed to compensate vehicle heat losses up to 10,000 Btu/hour as well as vehicle heat gains up to 4000 Btu/hour. A relatively simple EC/LSS radiator concept is utilized. The design recommendation is for dual coolant loops and radiators, with one pump operating and a second one in standby condition. The balance of the EC/LSS, as shown in Figure 4.20, is self-explanatory. No redundancy has been recommended for any of these subsystems.

Mode B — Backup Operating Mode

The system configuration for the backup operating mode is shown in Figure 4.21. The function of the atmospheric supply system is shown degraded in its capability for high-flow, short-period requirements (repressurization of cabin). In this mode, the high flow rate is dependent on the dual O_2 recovery unit capability. Otherwise, the system functions as it should in the primary mode.

The function of the water management system remains more or less unchanged, although only one of the two water reclamation units is in the loop, and it must work at full capacity.

Similarly, the atmospheric composition and temperature control system continues to function in a normal manner, with redundancy removed except for the humidity and air temperature control function.

The thermal control system is switched to the backup radiator and coolant loop and backup pump in order to afford possible maintenance and repair of the primary radiator and coolant loop and main pump. This too will not cause a degradation of performance.

The balance of the system functions continue in the same manner as designed for the primary operating mode.

Mode C – Emergency Operating Mode

The system configuration for the emergency operating mode is shown in Figure 4.22. It is designed to afford minimum crew life support requirements for 24-hour periods only. The atmospheric gas is supplied from one of the high-pressure storage tanks, provided that such a supply is available. If this should not be the case, one of the two O_2 recovery units would be operated at full capacity.

The water management system provides stored water at reduced pressure, and the water distribution is limited to restricted areas (example: command capsule).

Only one atmospheric composition and temperature control subsystem, with humidity air temperature control lacking redundancy, continues to operate. O_2 pressure is regulated by a single blower and may fall temporarily below nominal levels.

The thermal control unit capacity may be exceeded for short periods of time, with radiators and coolant loop water reaching the boiling point.

The food management system may loose the capacity for food preparation (heating). The waste management system function may be restricted to collection of waste within restricted areas only (example: command capsule). The crew supply subsystem function is reduced to storage only.

4.1.5 Conclusions and Recommendations on ECLSS

The results of the ECLSS indicate the feasibility and reason associated with the application of contemporary hardware to the extended space missions.

The results of the analysis are summarized in Table 4.14 which indicates that the system contribution to P_s can be raised from about 0.104 to over 0.999 with 146 spares. Table 4.15 presents support requirements and shows that only 213.6 pounds was required for the whole system.

This means that with the 146 M and R actions identified, there is less than one chance in one thousand that a spare may not be available if the ECLSS fails. This estimate is pessimistic in the sense that it does not consider the potential repair capability of the failed parts. Very few cases required redundancy to improve P_s either the component was satisfactory or it could be made so.

All of the identified M and R actions could be performed by a crewman either in zero or artificial gravity. None required use of a space suit. Many system design constraints were evolved since the identified M and R actions must be accommodated in the design concept.

The study indicated some rather important studies and test programs were required, both on earth and in earth orbit. The major problems were associated with the repair and/a replacement of components in liquid loops.

It is essential that a technique for breaking into these loops, reassembling the loop and restoring operation be developed, which will minimize the loss of fluids into the cabin atmosphere. Other areas of concern include the selection of the most appropriate components for the new system functions and the qualification of them as a function and system.

In Table 4.16 a classification code is developed for assessing the applicability of Apollo ECLSS hardware for the extended duration missions. The codes have been applied to an assessment of the baseline mission system requirements in Table 4.17.

Of the 13 ECLSS subsystem functions, all but three are considered already space rated and need only to demonstrate the life requirements. Of the remaining three, two are critical systems included in the P_s estimate.

<u>The Carbonation Cell</u> for CO_2 removal has only demonstrated feasibility, long term reliability and material compatibility are yet to be demonstrated. This is a critical system requiring immediate work.

<u>The Water Reclamation</u> function has been tested by several companies and feasibility has been demonstrated. But the effects of zero gravity and long-term operation are, as yet, unknown. Reclamation of urine, in particular, introduces peculiar problems which though resolved in the sense of feasibility, reliability is yet to be demonstrated. This system is also critical to P_s.

Table 4-14a Summary Analysis, ECLSS Availability

Group Number	Subsystem	Duty Cycle (hours)	Reliability	Number of Spares	Contribution to P_s	Spares Weight (lbs)
1.0	Radiator circuit	15,000	0.932	4	0.99995	8.8
2.0	Refrig. circuit		0.776	14	0.99983	20.3
3.0	Atmos. circuit		0.615	3	0.99996	6.2
4.0	Coolant circuit		0.866	4	0.99995	6.3
5.0	Humidity circuit		0.935	9	0.99989	31.2
6.0	Water reclamation		0.893	8	0.99991	24.8
7.0	Potable water storage		0.984	4	0.99995	1.1
8.0	Waste water storage		0.977	3	0.99996	0.9
9.0	Carbonization cell		0.862	17	0.99998	27.9
10.0	Cryo. oxygen supply		0.458	11	0.99987	22.5
11.0	High press. oxygen supply		0.856	8	0.99991	15.0
	Criticality I Totals		0.104	85	0.9991	165.0
12.0	Bosch reactor	1580	0.917	10	0.99952	23.9
13.0	Electrolysis cell	1580	0.981	8	0.99976	24.7
	Criticality II Totals		0.104	18	0.9993	48.6

Table 4.15. ECLSS Spares Requirements Tabulation (Criticality I)

System/Components		Number of Spares	Unit Weight	Spares Weight	Subsystem Totals
1.0	Radiator Circuit				
	Rad. flow control	4	2.0	8.0	
	Isolation valve	2	0.2	0.4	
	Check valve	2	0.2	0.4	
		8		8.8	8.8
2.0	Refrigerant Circuit				
	Refrigerant pump	2	1.5	3.0	
	Pump control	1	1.0	1.0	
	Evap. back pressure control	1	4.8	4.8	
	Cabin temperature control	2	0.9	1.8	
	Evap. water control	3	1.0	3.0	
	Heat exch. diverter	3	1.0	3.0	
	Refrig. temp. control valve	2	0.75	1.5	
	Refrig. shutoff valve	1	0.8	0.8	
	Refrig. check valve	2	0.1	0.2	
	Selector valve	2	0.6	1.2	
		19		20.3	20.3

Table 4. 15. ECLSS Spares Requirements Tabulation, (Criticality I) (Cont)

System/Components		Number of Spares	Unit Weight	Spares Weight	Subsystem Totals
3. 0	Atmosphere Circuit				
	Cabin Blower	3	0. 8	2. 4	
	Shutoff valve	5	0. 4	2. 0	
	Selector valve	3	0. 6	1. 8	
		11		6. 2	6. 2
4. 0	Coolant Circuit				
	Coolant pump	3	1. 5	4. 5	
	Selector valve	2	0. 6	1. 2	
	Shutoff valve	1	0. 4	0. 4	
	Check valve	2	0. 1	0. 2	
		8		6. 3	6. 3
5. 0	Humidity Circuit				
	Humidity control	2	0. 25	0. 5	
	Catalytic burner	1	10. 4	10. 4	
	Cat. burner control	3	3. 0	9. 0	
	Filter	1	0. 3	0. 3	
	Fan	2	1. 3	2. 6	
	Charcoal filter	1	1. 0	1. 0	
	Pump	2	2. 0	4. 0	

Table 4.15. ECLSS Spares Requirements Tabulation, (Criticality I) (Cont)

	System/Components	Number of Spares	Unit Weight	Spares Weight	Subsystem Totals
	Waste receiver	1	0.6	0.6	
	Selector valve	2	1.4	2.8	
		15		31.2	31.2
6.0	Water Reclamation				
	Condenser	1	5.15	5.2	
	Recuperator	1	5.15	5.2	
	Pyro. reactor	3	1.4	5.2	
	Evaporator	1	5.15	5.2	
	Pressure regulator	1	1.0	1.0	
	Heater control	3	0.9	2.7	
	Check valve	3	0.1	0.3	
		13		24.8	24.8
7.0	Potable Water Storage				
	Check valve	1	0.1	0.1	
	Shutoff valve	1	0.4	0.4	
	Pot. water outlet	1	0.6	0.6	
		3		1.1	1.1

Table 4.15. ECLSS Spares Requirements Tabulation, (Criticality I) (Cont)

	System/Components	Number of Spares	Unit Weight	Spares Weight	Subsystem Totals
8.0	Waste Water Storage				
	Check valve	1	0.1	0.1	
	Shutoff valve	2	0.4	0.8	
		3		0.9	0.9
9.0	Carbonation Cell				
	Filter	1	0.3	0.3	
	Fan	2	1.3	2.6	
	Water pump	2	2.0	4.0	
	CO_2 pump	3	5.0	15.0	
	Relief valve	2	2.4	4.8	
	Selector valve	2	0.6	1.2	
		12		27.9	27.9
10.0	Cryogenic Oxygen Supply				
	O_2 - Part. pressure control	1	0.8	0.8	
	Partial pressure sensor	1	0.2	0.2	
	Display	1	2.0	2.0	
	Regulator	1	1.0	1.0	
	Check valve	2	0.1	0.2	
	Shutoff valve	2	0.4	0.8	

Table 4.15. ECLSS Spares Requirements Tabulation, (Criticality I) (Cont)

	System/Components	Number of Spares	Unit Weight	Spares Weight	Subsystem Totals
	Selector valve	3	1.4	4.2	
	Heater control	4	0.9	3.6	
	Relief valve	3	2.4	7.2	
	Pressure transducer	5	0.5	2.5	
		23		22.5	22.5
11.0	High Pressure Oxygen Supply				
	Check valve	2	0.1	0.2	
	Shutoff valve	2	0.4	0.8	
	Selector valve	2	1.4	2.8	
	Relief valve	3	2.4	7.2	
	Regulator	1	1.0	1.0	
	Regulator	1	1.0	1.0	
	Regulator	2	1.0	2.0	
		13		15.0	15.0
		Criticality II			
12.0	Bosch Reactor Circuit				
	Bosch reactor	1	5.0	10.0	
	Heater control	1	0.9	0.9	
	Blower	1	3.5	3.5	

Table 4.15. ECLSS Spares Requirements Tabulation, (Criticality I) (Completed)

System/Components	Number of Spares	Unit Weight	Spares Weight	Subsystem Totals
H_2 pump	1	5.0	5.0	
Relief valve	1	2.4	2.4	
Shutoff valve	2	0.4	0.8	
Selector valve	2	1.4	2.8	
	10		23.9	23.9
13.0 Electrolysis Cell Circuit				
Anionic bed	1	8.0	8.0	
Cationic bed	1	8.0	8.0	
Relief valve	1	2.4	2.4	
Cell press. regulator	1	2.4	2.4	
Flow control	1	2.0	2.0	
Shutoff valve	1	0.4	0.4	
Check valve	1	0.1	0.1	
Selector valve	1	1.4	1.4	
	8		24.7	24.7
Criticality I Total	85	—	—	165.0
Criticality II Totals	103	—	—	213.6

The Bosch Reactor is required for emergency use only; however, since it is recommended, qualification is required. Although feasibility has been demonstrated in a test conducted for 25 days without interruption, little more has been accomplished. Therefore, both qualification in space and reliability for long life must be demonstrated.

Table 4. 16. EC/LSS Parts Applications Rating Key

Code	Prospect For Long-Duration Application
1	Not feasible at all
2	Feasible
2M	Feasible considering maintenance, or modification of respective Apollo hardware
3	Apparently acceptable as is
3M	Apparently acceptable considering maintenance, or modification of respective Apollo hardware

To preclude misinterpretation of the data in Table 4. 17 it should be noted that, although a considerable quantity of parts appear to be eligible for long-duration missions, the smaller group of ineligible items contains the major and significant components. Some redesign may be required to attain a satisfactory performance of those parts.

Table 4. 17. EC/LSS Parts Application Assessment

Parts Description	Potential Application*					
	1	2	2 M	3	3 M	**
CO_2 and odor absorber assembly	X					
Check valve		X				
Manual selector valve		X				
Demand pressure and relief valve		X				
Suit circuit return air check valve		X				
Debris trap					X	
Suit compressor						
No. 1	X					
No. 2	X					
Check valve		X				

Table 4. 17. EC/LSS Parts Application Assessment (Cont)

Parts Description	Potential Application*					
	1	2	2 M	3	3 M	**
Suit by-pass valve					X	
Cyclic accumulator valve assembly	X					
Cyclic accumulator control	X					
Cyclic accumulator O_2 warning sensor	X					
Signal conditioner						EL
Inverter valve		X				
Shutoff valve			X			
Check valve			X			
Glycol evaporator	X					
Glycol temperature control						EL
Temperature sensor					X	
Backpressure control valve			X			
Manual valve				X		
Glycol reservoir				X		
Backpressure control valve			X			
Backpressure temperature control						EL
Glycol temperature control valve			X			
Glycol temperature sensor					X	
Glycol temperature sensor					X	
Temperature sensor					X	
Glycol temperature sensor					X	
Water chiller				X		
Pressure transducer, compressor inlet				X		
Differential pressure transducer, suit compressor					X	
Pressure transducer, glycol pump outlet				X		
Pressure transducer, glycol accumulated quantity measurement				X		
Steam duct pressure switch (primary)			X			
Power supply temperature transducers					X	EL
O_2 pressure regulator assembly					X	
Emergency O_2 inflow control valve		X				
Cabin pressure regulator		X				
Tank pressure control and relief valve		X				
Check valve		X				
Flow transducer, oxygen supply				X		
Outlet pressure transducer, oxygen supply regulator				X		
Cabin pressure relief valve				X		
Potable water supply assembly				X		

Table 4.17. EC/LSS Parts Application Assessment (Cont)

Parts Description	Potential Application*					
	1	2	2 M	3	3 M	**
Suit flow limiter				X		
Suit hose connector assembly				X		
Check valves				X		
Shutoff valve				X		
Water tank pressure relief valve				X		
Water pressure relief valve				X		
Check valve				X		
Cabin temperature control valve	X					
Cabin heat exchanger				X		
Oxygen selector and check valve				X		
CO_2 and odor absorber removable container	X					
Suit compressor selector switch			X			
Switch, diverter valve			X			
Glycol pump selector switch			X			
Glycol shutoff valve				X		
Cabin temperature selector			X			
Cabin temperature anticipator			X			
Cabin temperature control						EL
Cabin temperature sensor			X			
Cabin blower closure				X		
Cabin blower selector switch			X			
Cabin recirculating blower	X					
Manual O_2 metering valve				X		
Main O_2 supply check valve			X			
Manual valve to O_2 surge tank				X		
Pressure regulator and relief valve			X			
Oxygen system filter			X			
Potable water tank			X			
Waste water tank			X			
Manual valve to potable water tank				X		
Post-landing ventilating blower		X				
Temperature sensor, gaseous, suit supply				X		
Pressure transducer, steam duct (primary)				X		
Temperature sensor (steam duct)				X		
Space radiator outlet temperature transducer				X		
Command module cabin atmospheric temperature sensor				X		
Command module cabin total pressure transducer				X		
Waste water tank quantity measurement pressure						

Table 4. 17. EC/LSS Parts Application Assessment (Completed)

Parts Description	Potential Application*					
	1	2	2 M	3	3 M	**
transducer				X		
Potable water tank quantity measurement pressure transducer				X		
Sensor amplifier(s)						EL
CO_2 sensor	X					
Steam duct heater		X				
Urine nozzle heater		X				
Manual backup valve				X		
Battery vent valve				X		
Vacuum cleaner				X		
Urine nozzle				X		
Fecal canister assembly				X		
Flex hose				X		
Pump package	X					
Suit circuit cooling	X					
Glycol evaporator		X				
Glycol temperature sensor				X		
Back pressure control valve		X				
Back pressure temperature control						EL
Water control valve		X				
Oxygen shutoff valve		X				
Water glycol check valve		X				
Glycol pump selector switch	X					
Inverter valve				X		
Command module LM pressurization valve				X		
Pressure equalization valve				X		
LM pressure gauge				X		
Command module hatch vent valve				X		
Post-landing ventilating outlet valve				X		
Pressure transducer, steam duct				X		
Check valve				X		
Glycol manual metering valve				X		

*See Rating Key, Table 4. 16.
**Wholly electrical item

4.2 ATTITUDE AND STABILITY CONTROL SYSTEM (A and SCS)

Note: The data used herein was supplied by the Honeywell Corporation of Minneapolis, Minn. Ref. 4.4.

4.2.1 A and SC System Functions

By definition, the A and SCS functions are required during the zero gravity mode only. The gravity control systems are included under a separate heading. In the zero gravity mode, the A and SCS is required to function so as to maintain spacecraft stability in pitch, roll and yaw and to facilitate the acquisition and maintenance of any attitude with respect to its line of travel or any other selected reference. The functions required for the planetary mission are the same as those required for the lunar mission. Mission success and crew safety is therefore dependent on accomplishment of the following three critical A and SCS functions:

1. Attitude Hold Vehicle Maneuver Functions

2. Thrusting Function

3. Entry Function (earth entry module only)

Except where noted, these functions are required for all separable modules. This includes the Mission Module and Earth Entry Module for the baseline flyby spacecraft.

These functions are to be performed using Apollo hardware as follows:

1. During thrusting, the A and SCS controls the vehicle attitude by positioning the gimbals of the thrusting engine in pitch and yaw. Roll attitude is controlled by A and SCS operation of the roll reaction jets. The A and SCS provides the on-off control of the thrusting engine.

2. During non-thrusting periods, vehicle attitude and rotational maneuvers are performed by SCS control of reaction jets in all three axes.

3. The A and SCS displays provide a presentation of vehicle attitude, attitude error and maneuvering rate. In addition, fuel pressure, gimbal position, attitude set and a redundant roll angle during entry are displayed.

4. Redundant body-mounted gyro assemblies sense vehicle rate and attitude used for attitude hold-vehicle maneuver, and display purposes.

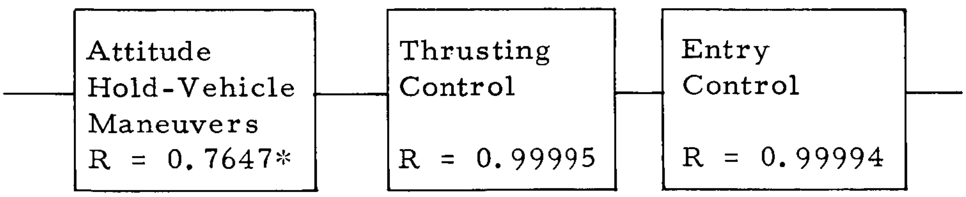
The Apollo SCS, with the addition of an accelerometer and g-display, is capable of performing all the required critical functions for the baseline mission. Achievement of the specified mission reliability and safety requires in-flight maintenance. The maintenance activities consist of replacing the identified components essential to successful completion of the critical functions as they fail, and if they fail. A complete system description and schematic may be found in Reference 1. 1.

4. 2. 2 Mission Reliability Assessment

The system function reliabilities were estimated first on the basis of no maintenance, using the Apollo data and taken from Reference 4. 4 and 4. 5. In the zero gravity mode, the A and SCS provides the functions previously described and the reliability estimates without M and R are as reflected in the logic of Figure 4. 23.

As taken from Volume I, the estimated duty cycles are as follows:

1.	Attitude hold-vehicle maneuver	904. 0 hrs
2.	Thrusting maneuvers	4. 0 hrs
3.	Entry control (Earth Entry Module)	0. 4 hrs



*Automatic Mode only; inclusion of manual modes raises to at least 0. 8793

Figure 4. 23. Top Level A and SCS Reliability Logic

From this figure, it is evident that the predominent weak link is the attitude hold-vehicle maneuver functions. These are explored in detail in the subsequent analysis.

Reliability calculations were made assuming that the zero g configuration functions were performed in time sequence, i. e., the 904 hours of attitude hold and vehicle maneuver would be performed before the thrusting function and the thrusting function would be completed prior to entry. While it is common knowledge that vehicle thrusting occurs at intervals after achieving earth orbit and before entry, the assumption greatly simplified calculations without any significant effects on the accuracy of results. It was also assumed that the only A and SCS components powered were those components required for the particular function being performed.

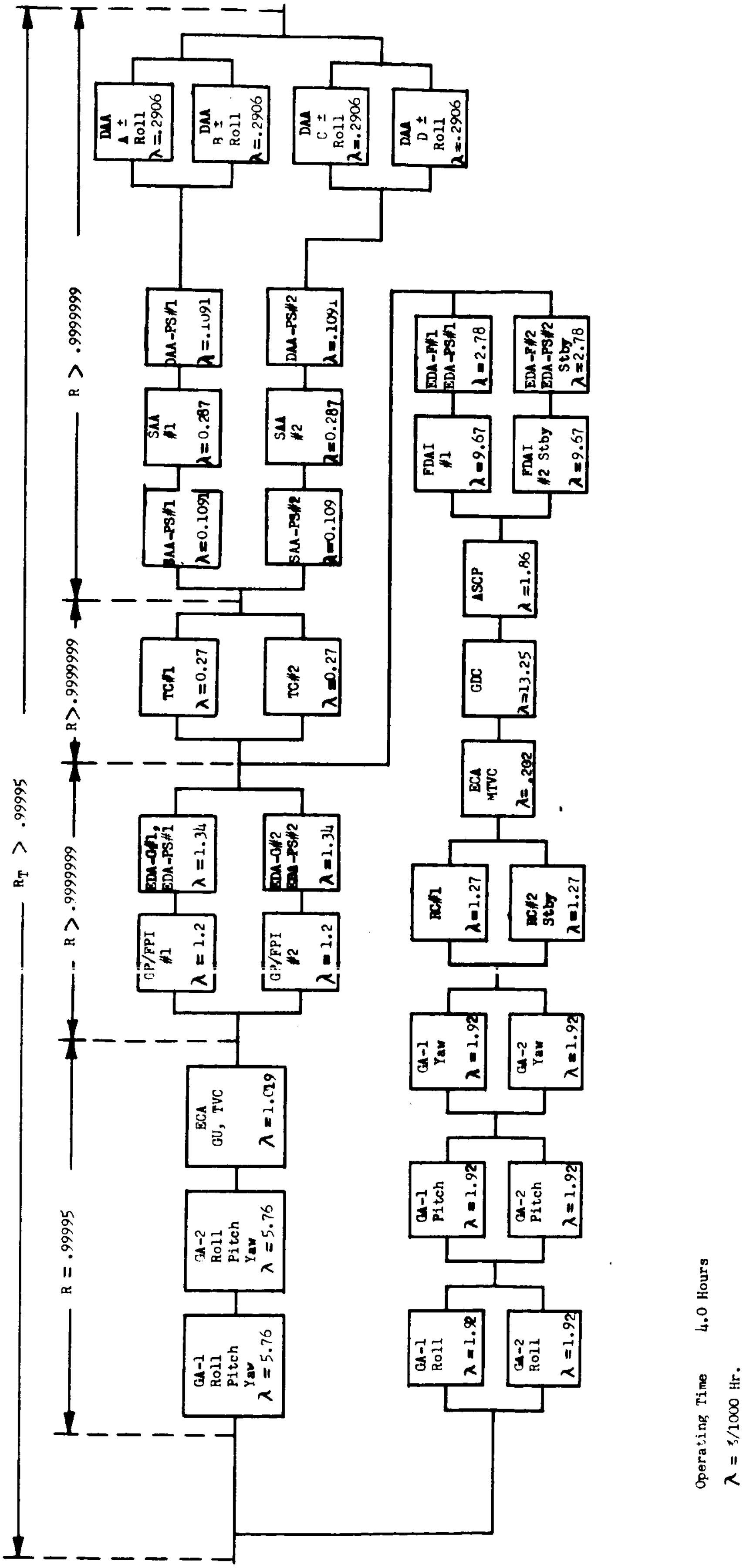
The probability of success for the entire mission was developed from the product of the individual success probabilities of the five critical functions, where these individual probabilities were conditioned to reflect the probability that the equipment would be properly operating at the beginning of each function. Because of the relatively short operating times for the thrusting and entry functions, these functions do not significantly affect the reliability of the A and SCS. Therefore, the availability improvement study considered only the reliability models associated with the attitude hold-vehicle maneuver functions.

The failure rates utilized in the reliability logic diagrams are the predicted failure rates of identical or similar subassemblies in the Apollo system. These represent the latest and best source of space rated hardware reliability data.

The probability of completing the zero-g mode of operation was determined by combining the probabilities of completing the thrusting function, entry function and the attitude hold-vehicle maneuver function. The analysis of each mode is presented below.

1. The Thrusting Function

The reliability block diagram for the Apollo A and SCS used for the thrusting function is shown in Figure 4. 24. A worst-case reliability analysis was made to determine if the reliability during this function would affect the overall reliability of the A and SCS capabilities. Although there are two modes of operation, it was assumed that both set of gyros had to operate and were needed. With this pessimistic assumption, the reliability of the Apollo system for the thrusting function on the baseline mission is 0. 99995. Since the actual reliability, considering both modes of operation, will be greater than 0. 99995, and since this is significantly larger than the mission reliability goal, this function will require no maintenance for the planetary mission.



Operating Time 4.0 Hours

λ = f/1000 Hr.

Figure 4.24. Functional Logic-Thrusting Function

2. Entry Function

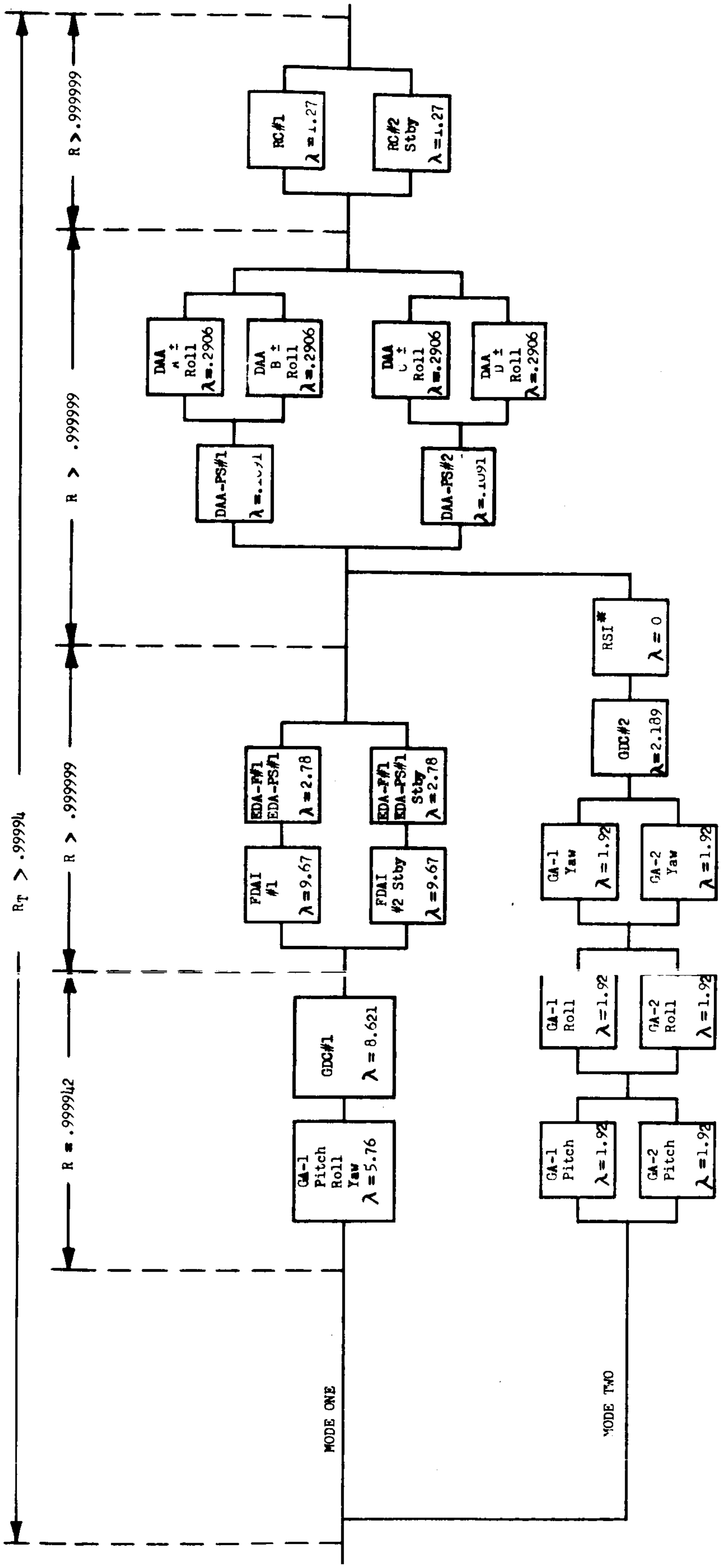
The reliability block diagram for the Apollo System as used for the Entry function is shown in Figure 4.25. The SCS has two modes of operation during the entry function and is only required for the EEM during earth entry. To simplify the calculations and consider the worst-case condition, it was assumed that the most complex mode, requiring the use of the gyros (GA-1) and the FDAI was required. With this assumption, the reliability using Apollo hardware was calculated to be at least 0.99994 for the reentry function. With this high reliability, the probability of requiring a maintenance action is negligible.

3. Attitude Hold-Vehicle Maneuver Function

The proposed system, as for Apollo, provides four methods of performing vehicle Attitude Hold Maneuver functions. The modes of command, in order of increasing hardware complexity, are: (1) direct command, (2) acceleration command, (3) minimum impulse, and (4) automatic command. Direct, acceleration and minimum impulse commands are initiated by the astronaut via the rotation control. The automatic commands are derived from the rate and position sensing gyros and provide automatic attitude hold using closed loop operation.

The reliability block diagrams for the automatic command mode is shown in Figure 4.26. Figure 4.27 shows the block diagram for the direct command, acceleration command, and minimum impulse command. Many of the components used in the direct mode are also common to the other three modes. In addition to these common components, the acceleration command mode requires the jet driver amplifiers, the minimum impulse mode requires the jet driver amplifiers, the reaction jet control and power supply A of the Electronics Control Assembly (ECA), and the automatic mode requires the jet driver amplifiers and the entire ECA.

Since the direct command mode requires less equipment, the highest probability of having one of the four modes operable is equal to the probability of the direct command mode being operable which is only 0.879. The same reasoning can be used to show that the probability of having the acceleration command, minimum impulse, or automatic command operable is equal to the probability of having the acceleration command operable which is also about 0.879. The probability of having either the minimum impulse or automatic command operable is equal to the probability of having the minimum impulse mode operable which is only 0.859.



Operating Time: 0.4 Hours

λ = %/1000 Hr.

*RSI = Roll Stability Indicator (Non-SCS Component)

Figure 4.25. Functional Logic - Entry Function

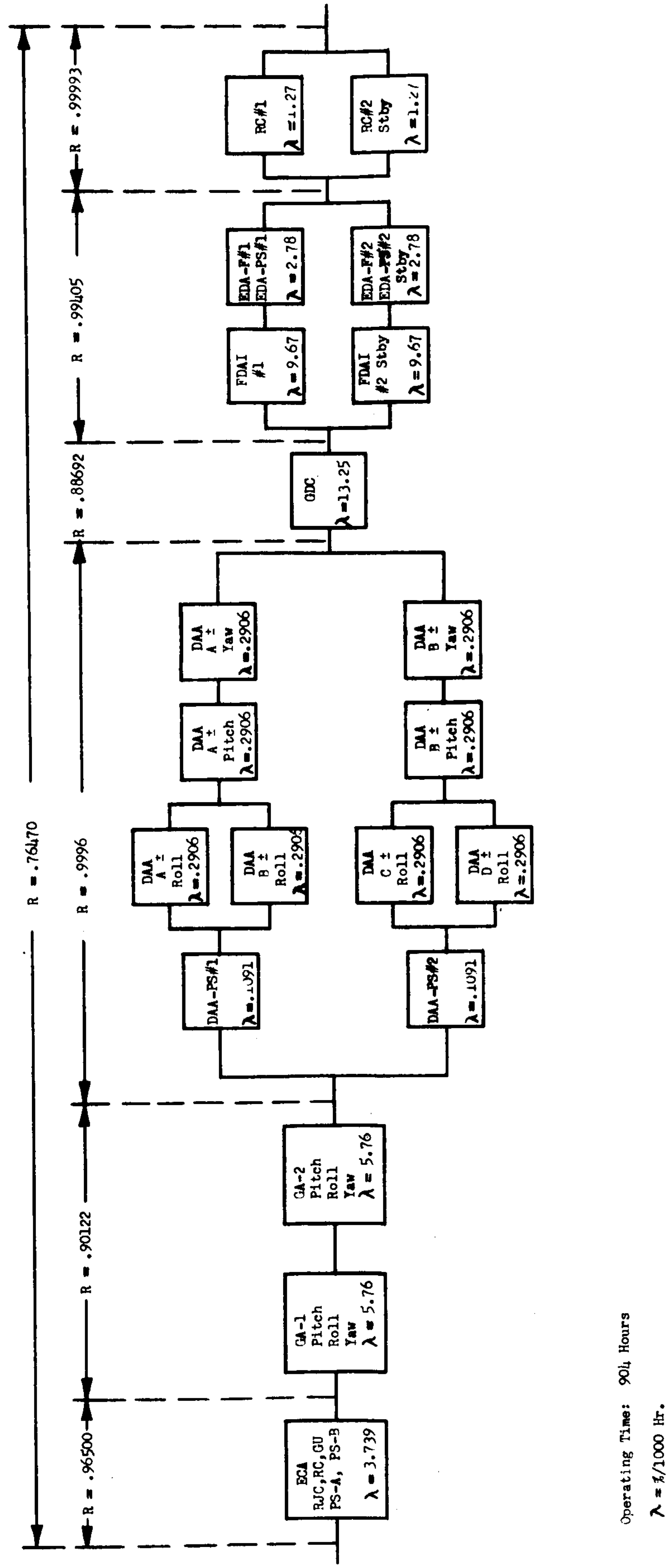


Figure 4.26. Reliability Block Diagram - Automatic Command Mode Attitude Hold-Vehicle Maneuver

Table 4.18. Component Level Sparing Alternatives Attitude Hold-Vehicle Maneuver Function

Mode of Operation	Spares Required					
	1	2	3	4	5	6
	No Spares	1 - GDC 1 - FDAI 1 - EDA	1 - GDC 1 - FDAI 1 - EDA 1 - ECA	1 - GDC 1 - FDAI 1 - EDA 1 - ECA 1 - GA	1 - GDC 1 - FDAI 1 - EDA 1 - ECA 1 - GA	2 - GDC 1 - FDAI 1 - EDA 1 - ECA 1 - GA
Automatic Command	0.76470	0.86140	0.89218	0.98710	0.99219	0.99345
Minimum Impulse	0.85883	0.96743	0.99031	0.99272	0.99277	0.99907
Acceleration Command	0.87930	0.99049	0.99058	0.49295	0.99301	0.99934
Direct Command	0.87934	0.99053	0.99062	0.99299	0.99305	0.99938

The results of the analysis show that the reliability for the mission module attitude and stability control function is inadequate for the planetary mission without maintenance or repair of the stability control function.

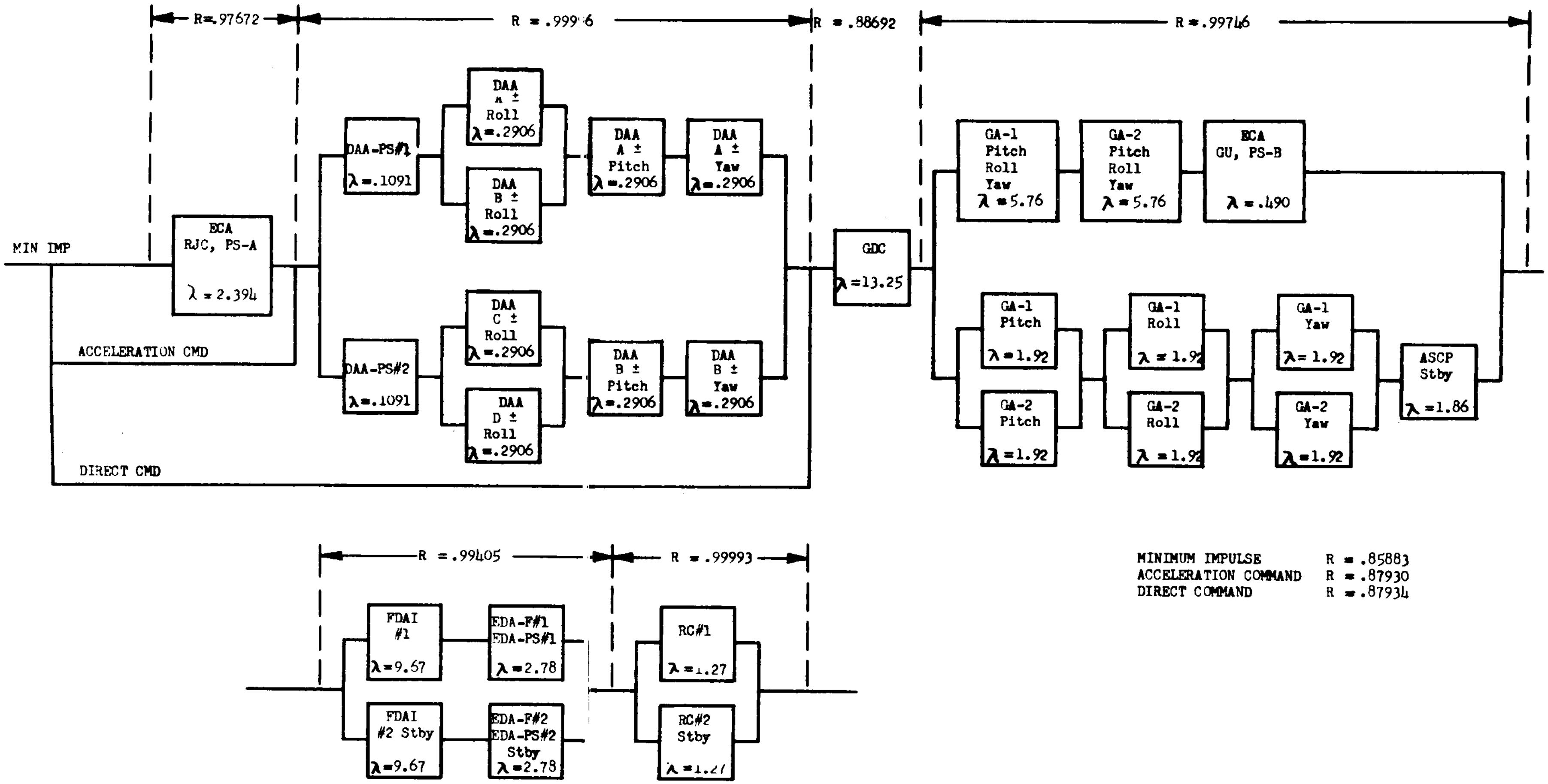
4.2.3 Availability Analysis

The relatively long operating time of the Manned Mars Flyby mission necessitates improvement in the baseline system reliabilities. The required level of safety was achieved by sparing the baseline system. System weight, volume and maintainability requirements were first established based on sparing at the component level. A second iteration investigated the weight, volume and maintainability advantages of sparing at the module level to achieve equal mission safety. In both cases maintenance was possible and spares were added to the system, as required, to eliminate weak links, equalize risk and to lower the probability of a failure (and no spare) for each mode of vehicle maneuver and attitude hold to a minimum of 0.99. The resulting contributions to P_s was less than 0.99938 with 6 spares.

1. Component Level Sparing

Examination of Figure 4.27 shows the weak links in the attitude hold vehicle maneuver reliability-logic diagram are the GDC, FDAI/EDA, the gyro assemblies and the ECA. These components were spared as shown in Table 4.2-1. Column 1 presents the reliabilities of the baseline attitude hold-vehicle maneuver function as determined in the preceeding section. Column 2 shows the effect of adding a spare GDC, FDAI and EDA. The spare GDC increased the GDC contribution to P_s from 0.99405 to 0.99977. The effect of sparing on the various modes can be determined by comparing Column 1 with Column 2. Since the weakest links in the direct and acceleration command mode are the GDC, FDAI and the EDA, these modes benefited most from sparing these components. The addition of a spare GDC, FDAI and EDA brought the direct acceleration command modes within the objective, however, the minimum impulse and automatic command modes require additional spares. Referring to Figure 4.2-5, the weak link in the minimum impulse mode is the ECA. The addition of a spare ECA increases the P_s for that function from 0.9762 to 0.99468. This spares level increases the minimum impulse mode P_s to 0.99031. The level of spares and associated maintenance considered to this point would only increase the safety or P_s of the automatic mode of operation to 0.89218. The weak links left are the gyros and the GDC even through a spare GDC is already incorporated.

If we consider adding one spare gyro package to the Apollo concept which consists of three gyros and associated electronics along with the redundant GDC, a spare FDAI/EDA and a spare ECA, the P_s for the automatic mode of operation would increase to 0.98710. A second spare gyro package is required to increase the P_s to 0.99219. The desired level of



MINIMUM IMPULSE	R = .85883
ACCELERATION COMMAND	R = .87930
DIRECT COMMAND	R = .87934

Operating Time: 904 Hours

λ = %/1000 Hr.

Figure 4.27. Reliability Block Diagram Minimum Impulse, Acceleration and Direct Command Modes Attitude Hold-Vehicle Maneuver

safety can also be achieved by adding one spare gyro package and a second spare GDC to the spares level mentioned previously. This results in a A and SCS contribution to P_s of 0.99345.

The results of the sparing analysis at the Component level is presented in Table 4.2-1. It indicates that six spare components were required to achieve better than 0.9994 contribution to the probability of no failure which cannot be restored (or 0.9994 contribution to P_{CS}); and a 0.9935 probability that the automatic mode will be kept functional throughout the mission as required.

2. Module Level Sparing

Sparing on the module level was investigated for potential weight and volume benefits and to permit the most efficient use of the equipment carried on board the spacecraft. The following discussion identifies the A and SCS components that are modularized and gives a description of the recommended modularization techniques for each component, Table 4.2-2 summarizes the spares required for the module level M and R concept.

The GDC. The proposed module sparing for the GDC is shown in Figure 4.2-5. Advantage was taken of the commonality of the voltage-to-frequency converters and the resolver assemblies for weight and volume improvements. The module sparing approach shown in Figure 4.2-5 increases the reliability of the GDC from 0.88692 to 0.99806. This is a significant increase over the reliabillty of 0.99335 obtained using one complete GDC spare at the component level.

The FDAI/EDA. The proposed module sparing for the FDAI and EDA is shown in Figure 4.27. The physical characteristics of the Apollo FDAI impede sparing this device at the module level. Maintainability considerations make it more desirable to spare the FDAI at the component level. The EDA electronics and power supplies can be packaged individually or shown in Figure 4.27. The existing Apollo EDA Channel No. 1 and Channel No. 2 electronics are interchangeable as are power supply No. 1 and power supply No. 2. The spare FDAI and the EDA interchangeability feature assures a FDAI/EDA P_s of 0.99968. This is only slightly less than the P_s of 0.99977 obtained by using one spare FDAI and one spare EDA at the component level but the weight advantage makes it more advantageous.

The ECA. The proposed module sparing of the ECA is shown in Figure 4.2-8. The P_s of the ECA using module replacement is 0.99987 as compared to a P_s of 0.99938 by sparing the complete ECA package. By

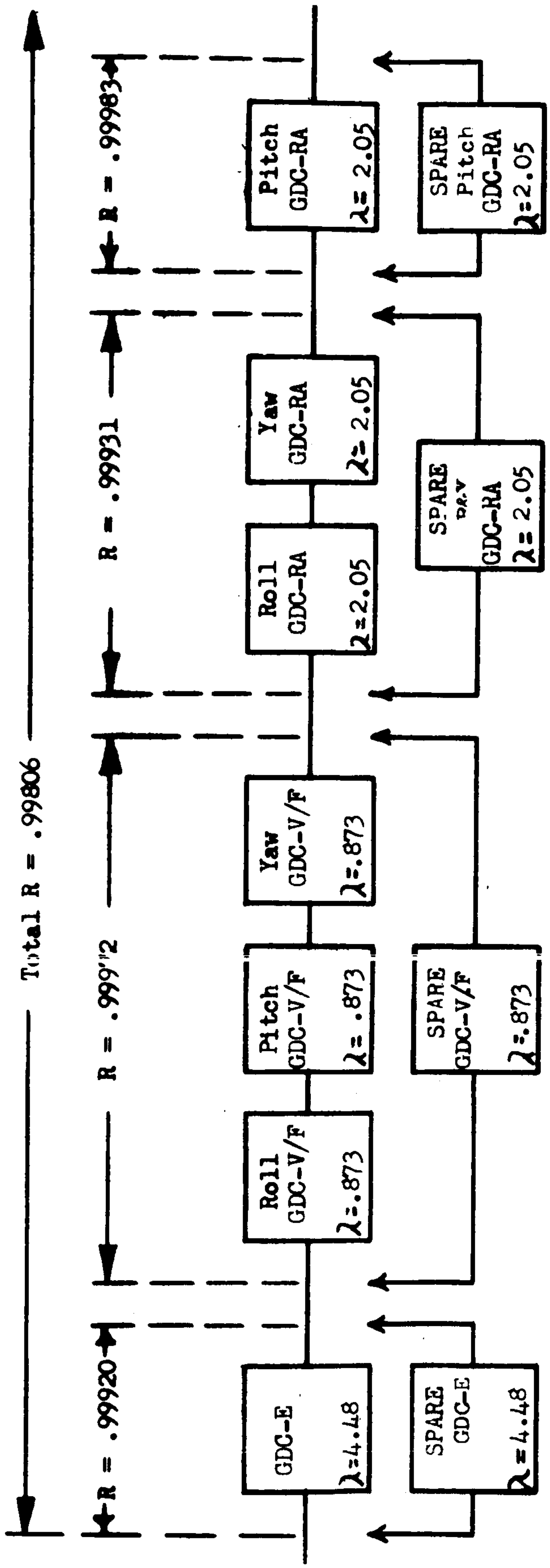


Figure 4.28. GDC Module Sparing

sparing at the module level, it is possible to eliminate one FJC axis spare, one power supply spare and one axis RC spare and still end up with a higher P_s than by sparing the complete ECA package.

The Gyro Assembly. The proposed module sparing of the gyro assemblies is shown in Figure 4.29. The modularized gyro packages each contain a gyro and associated electronics that is capable of operating in the attitude mode as GA-1 or in the rate mode as GA-2. The P_s of the gyro assemblies using module level replacement is 0.99457. The P_s using component level replacement is the same since two gyro failures at either the module or component level will result in the loss of the automatic mode of vehicle attitude hold. The Apollo concept has two gyro packages, each containing three gyros and three sets of gyro electronics. During the automatic mode of attitude hold the two packages operate in series (reference Figure 4.29). If component sparing is used, a spare gyro package has to be carried consisting of three gyros and electronics. If one gyro in one of the primary packages fails, the complete package will have to be replaced by the spare gyro package. If a second gyro fails, there will be a system failure because there will not be two operating gyro packages, although there are still seven good gyros on board, and only six are required. The system will be failed because there is at least one failed gyro in each of the packages.

Using the module sparing approach, the same P_s can be obtained by carrying one gyro spare and its electronics. In this case, if a gyro fails in either of the two packages, the failed gyro will be replaced by the spare gyro. Hence, using the module sparing approach results in the same P_s but more efficient use of the weight allocation and conversly, higher safety and P_s is achievable for the given weight allocation.

4.2.4 Achievements through Availability

The results of this study demonstrate the Apollo designs can fulfill all of the A and SCS functional needs for the planetary missions. The 700 day mission duration objectives can be met through the addition of about 19 pounds of spares at the module level or 40 pounds of spares at the component level. Sparing at the module level requires about 1,185 cubic inches less than at the component level. However, sparing at the module level would introduce a requirement for some repackaging while at the component level the design could remain virtually unchanged.

The recommended system utilizes sparing at the module level in accordance with Figures 4.28 through 4.31. Table 4.19 shows the spares required to obtain the given mission P_s for each mode of A and SCS operation. The spares and P_s using component-level sparing are also shown for comparison purposes. The system weight and volume data in Table 4.2-4 shows

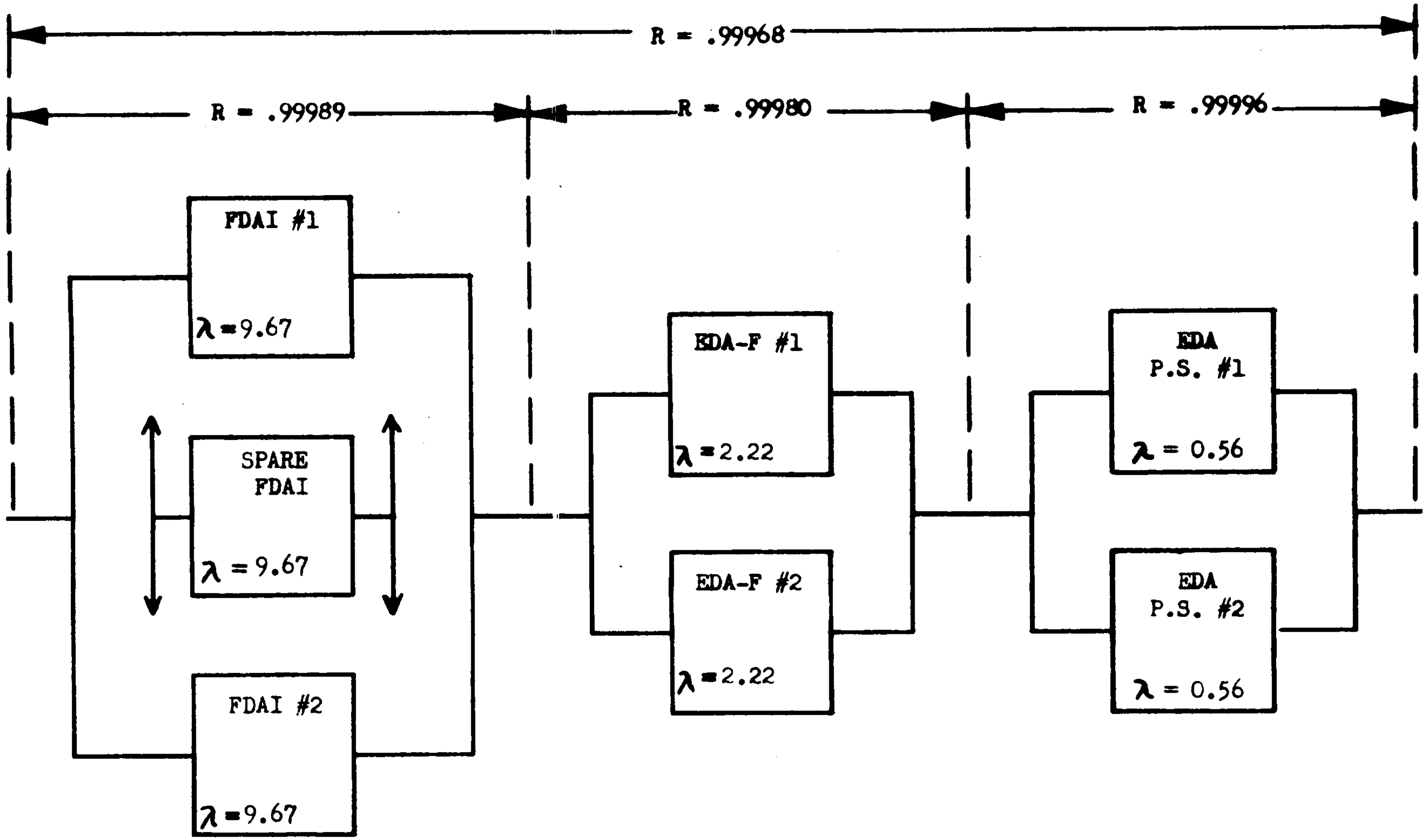
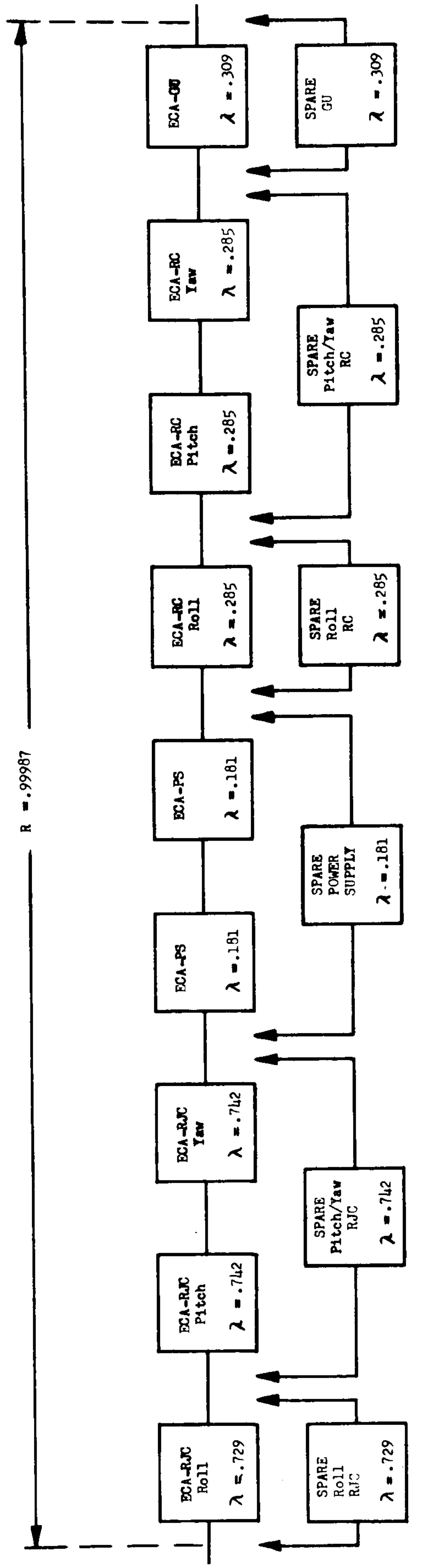


Figure 4.29. FDAI Module Sparing



λ = %/1000 Hr.

Figure 4.30. ECA Module Sparing

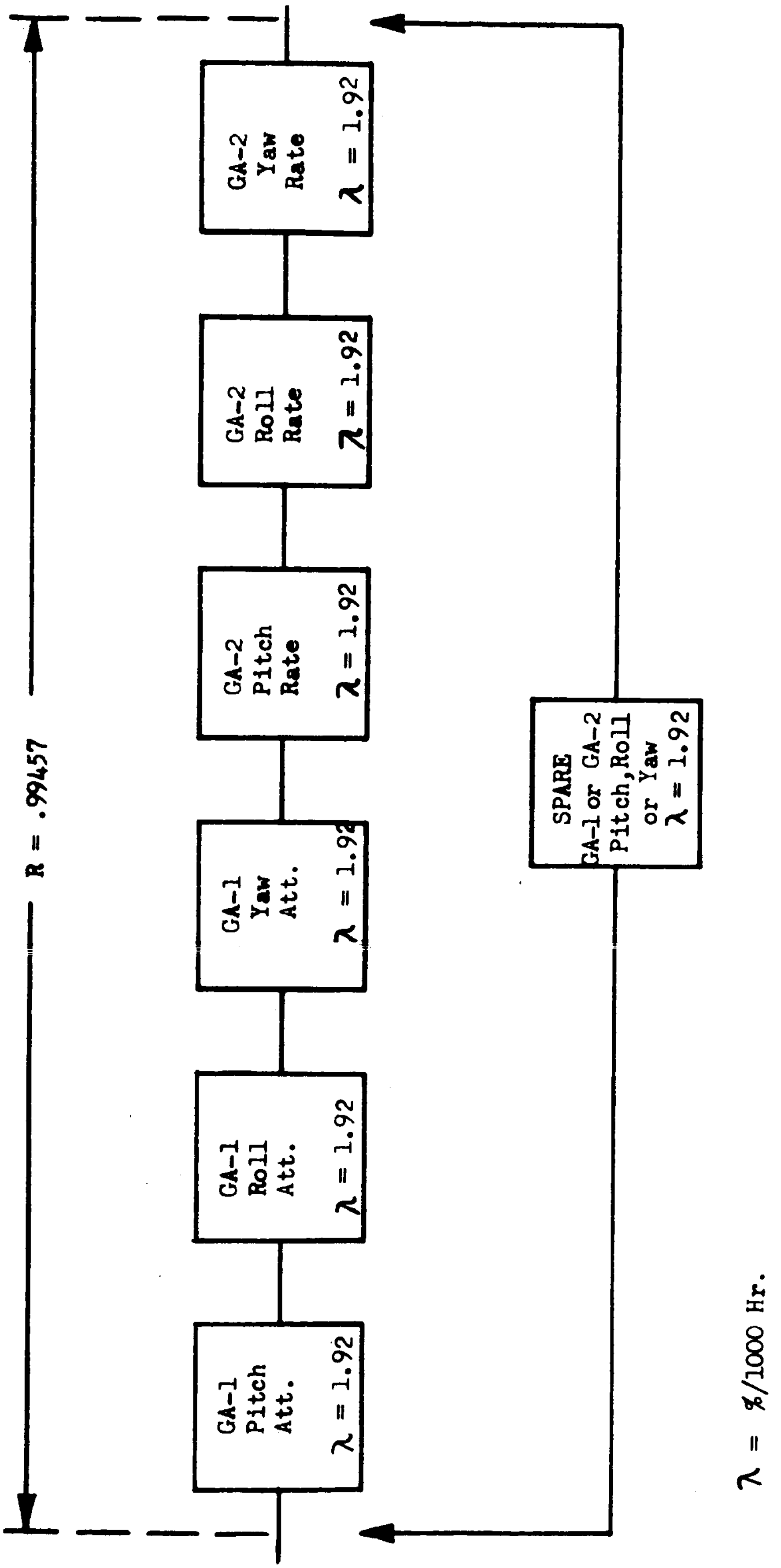


Figure 4.31. Gyro Assembly Module Sparing

Figure 4.32. Photo, A&SCS Electronics, Open for M&R

Table 4.19. Sparing Requirements, Module Level Maintenance Concept

No. Req.	Spare Required	Weight Lbs.	Volume Cubic-Inches
1	EDA/Power supply module*	7.5	360
1	FDAI	8.9	350
1	Attitude/rate gyro module	7	220
1	Roll RJC module	0.92	30
1	Pitch/yaw RJC module	0.92	30
1	Power supply module	0.61	32
1	Roll rotation control elect. module	0.41	21
1	Pitch/yaw rotation cont. electronic module	0.72	30
1	Spare gyro uncage logic module	0.5	15
1	V/F converter	0.98	30.0
2	Resolver assemblies	5.2/10.4	73.0/146
1	Electronic module	6.0	228.0
13	Totals	19.62	1462

the prime advantage of sparing at the module level. The data considers the total weight and volume of the system components to achieve the P_S values given in Table 4.20. Weight and volume of the cables required to interconnect the system components was not considered.

4.2.5 Maintainability Inferences

The maintainability considerations are approximately the same for both the component and module level sparing concepts of in-flight maintenance. This assumes that the module packaging is consistent with present Apollo

Table 4.20. Extended-Mission Redundancy and Reliabilities

	Spares Required	
	Module Sparing*	Component Sparing
A and SCS Mode of operation	4 for - GDC (See Fig. 4.28) 1 - FDAI/EDA Component 6 for - ECA (See Fig. 4.30) 1 - Gyro 2 - AE/GD 2.- Accelerometers	1 - GDC 1 - FDAI/EDA 1 - ECA 2 - Gyro Assy 2 - AE/GD 2 - Accelerometers
Automatic command	0.99130	0.99136
Minimum impulse	0.99662	0.99194
Acceleration command	0.99673	0.99218
Direct command	0.99677	0.99222

*Recommended Sparing Concept

Table 4.21. SCS Weight and Volume Summary Module-Level Sparing vs. Component Level Sparing

SCS Components	Weight - lbs		Volume - cu-in	
	Comp Sparing	Module Sparing	Comp Sparing	Module Sparing
1. Spared Components				
GDC	49.40	41.48	1,390	1,251
EDA/FDAI	73.60	63.70	2,636	2,380
ECA	32.20	23.58	1,830	1,720
Gyro assy	90.00	63.00	2,440	1,760
Accel - AE/GD	6.0	6.0	45	45
Total - Spared components	251.20	134.76	8,341	7,156
2. Non-Spared Components				
2 - Rotation controls	13.9	13.9	270	270
RJ/EC	19.6	19.6	850	850
Trans. control	5.6	5.6	97	97
ASCP	3.3	3.3	74	74
TVSA	12.4	12.4	450	450
GP/FPI	2.85	2.85	77	77
Total - Non-spared components	75.65	75.65	1,818	1,818
SCS Total	326.85	210.41	10,159	8,974

component packaging concept. Regardless of the sparing concept, the in-flight replaceable units require the following considerations:

1. The units requiring physical replacement must have fasteners compatible with zero-g environment.

2. The units requiring electrical replacement should have connectors that are readily connected and disconnected in a zero g shirtsleeve environment, preferably without tools.

3. Where possible, spare units should be located adjacent to the replaceable unit to preclude physical replacement.

4. Additional means of protection should be considered for cable connectors exposed during repair, stowage and handling.

5. Fault isolation to a defective unit must be included.

The maintainability of the A and SCS for extended missions will vary considerably depending on the extent the above considerations are reflected in the design. Preliminary redesign investigations indicate that the present Apollo components can be properly maintained within the mission constraints. The system has existing parallel redundancy for each mission critical function. This feature eliminates the possibility of exceeding the minimum downtime allowable. Figure 4.30 is included to verify maintainability in its present (Apollo) form.

The use of module level sparing would enhance the A and SCS maintainability since less of the system function would be inoperable during repair. Also the smaller modules would facilitate handling, stowage and replacement.

4.2.6 Recommendations for Future Studies

During the course of this study specific potential problem areas were identified that were beyond the scope of this study. These problems must be resolved by future studies. These include:

1. Required Time Between Maintenance. The Apollo components, except the gyros, are designed to operate for 1400 hours after a shelf life of 3 years. The gyro assembly is a limited-lifetime item and requires checkout and recalibration, if necessary, every 6 months. The gyros, FDAI, GDC, ASCP, GP/FPI and G-display will require special maintenance attention for the planetary mission. These devices have bearings that will probably require special lubrication schedules to insure proper operation during a 700-day mission. Component redesign using state-of-the-art material might extend the time between maintenance to be compatible with the extended mission times. The

mean time between maintenance of the EDA, ECA, TVSA, RJ/EC and accelerometer should not present any problems.

2. Spares Checkout Capability. Consideration must be given to providing on-board facilities to insure the spares will be operable before they are required. Maximum use should be made of the redundant SCS channels for this purpose to conserve volume and weight.

3. Detailed Redesign Investigation. The redesign investigation performed during this study was sufficient to provide reliable weight and volume estimates. A detailed redesign effort is required to insure that each component is built to facilitate installation, removal stowage and handling. The redesign consideration must be given to those components that require recalibration and lubrication during flight.

4.3 REACTION CONTROL ENGINES

Note that much of the data used in this section was provided by the Marquardt Corporation in Reference 4.6.

4.3.1 Functional Description

The Reaction Control Engines (RCS) are required to perform several functions on the manned planetary spacecraft. In each case, the R-4D Marquardt engine used for the Apollo Service Module was found to be adequate. The functions required are:

1. Spacecraft attitude change and maintenance, zero gravity mode

2. Spinup and Despin, artificial gravity mode

3. Spin plane precession, artificial gravity mode

The long duration mission exposes these engines to the space environment for up to two years, require long burn times of over an hour each time (for spin/despin) and requires in some cases, tens of thousands of ignitions. These requirements far exceed those imposed by Apollo; but because of the other uses the R-4D has been employed, much data is available to provide assurance in its capability to meet the mission requirements. A typical schematic diagram is given in Figure 4.33.

4.3.2 Reliability Assessment

The reliability of the R-40 engine has been assessed against each of the required functions and found to be adequate in all respects to meet the baseline mission requirements. From Table 4.22 it can be seen that in the most stringent function, that of stability control, the reliability in this function exceeds 0.99975; without any form of maintenance.

A detailed analysis of failure probability, failure mode and effects, and a critical items list for the Model R-4D-4 on the baseline mission was made, (see Reference 4.3-1). The reliability logic block diagram is presented in Figure 4.34 for the thrust chamber assembly (basic engine) and in Figure 4.35 for the Model R-4D-4 engine; the latter includes the accessory equipment consisting of heater, redundant thermostat switches, and arc suppression diodes.

The results of a reliability analysis is presented in Figure 4.35. It was assumed that the burn life and structural capabilities of the engine were

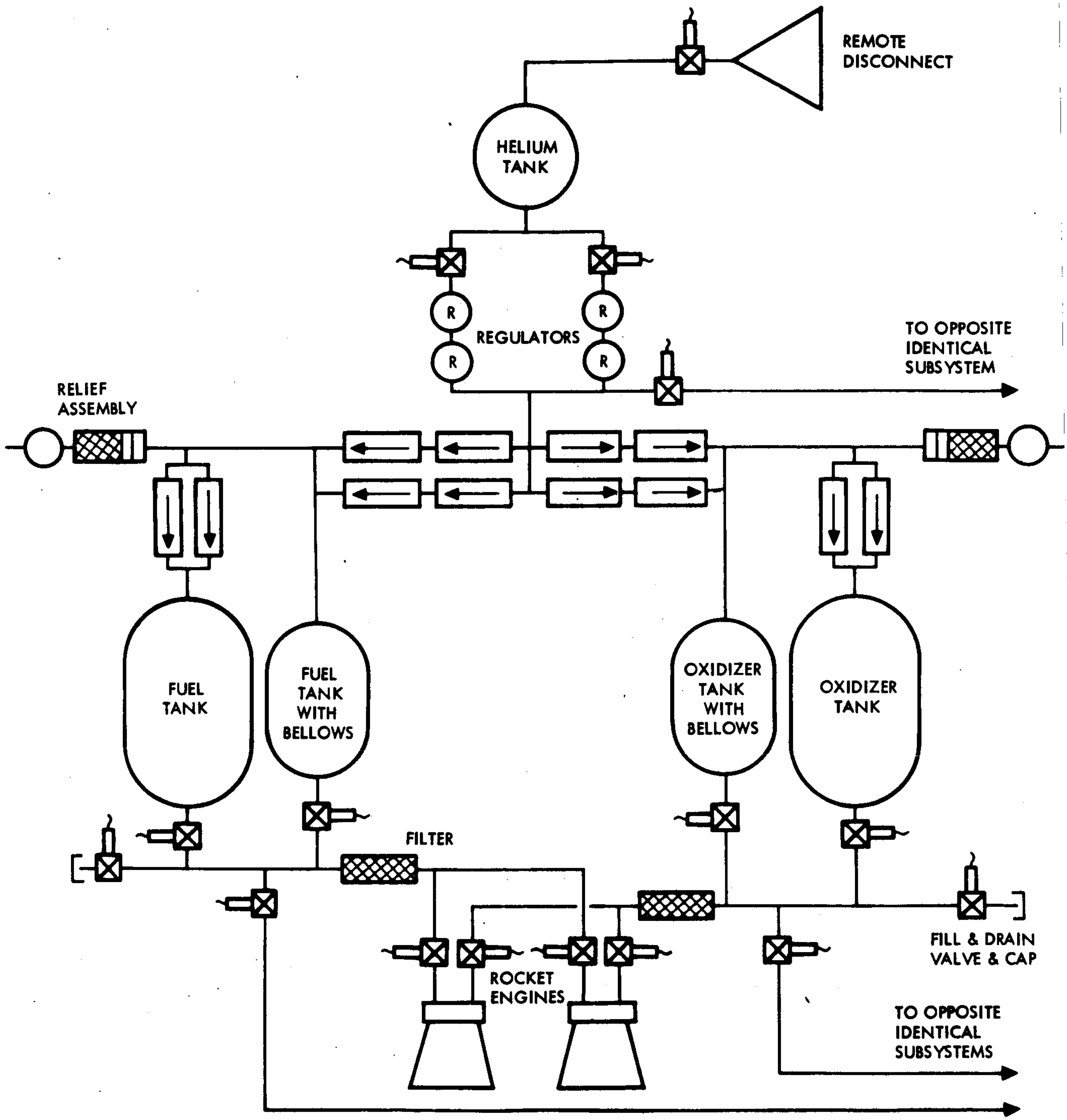


Figure 4.33. Spin Engine Subsystem Schematic

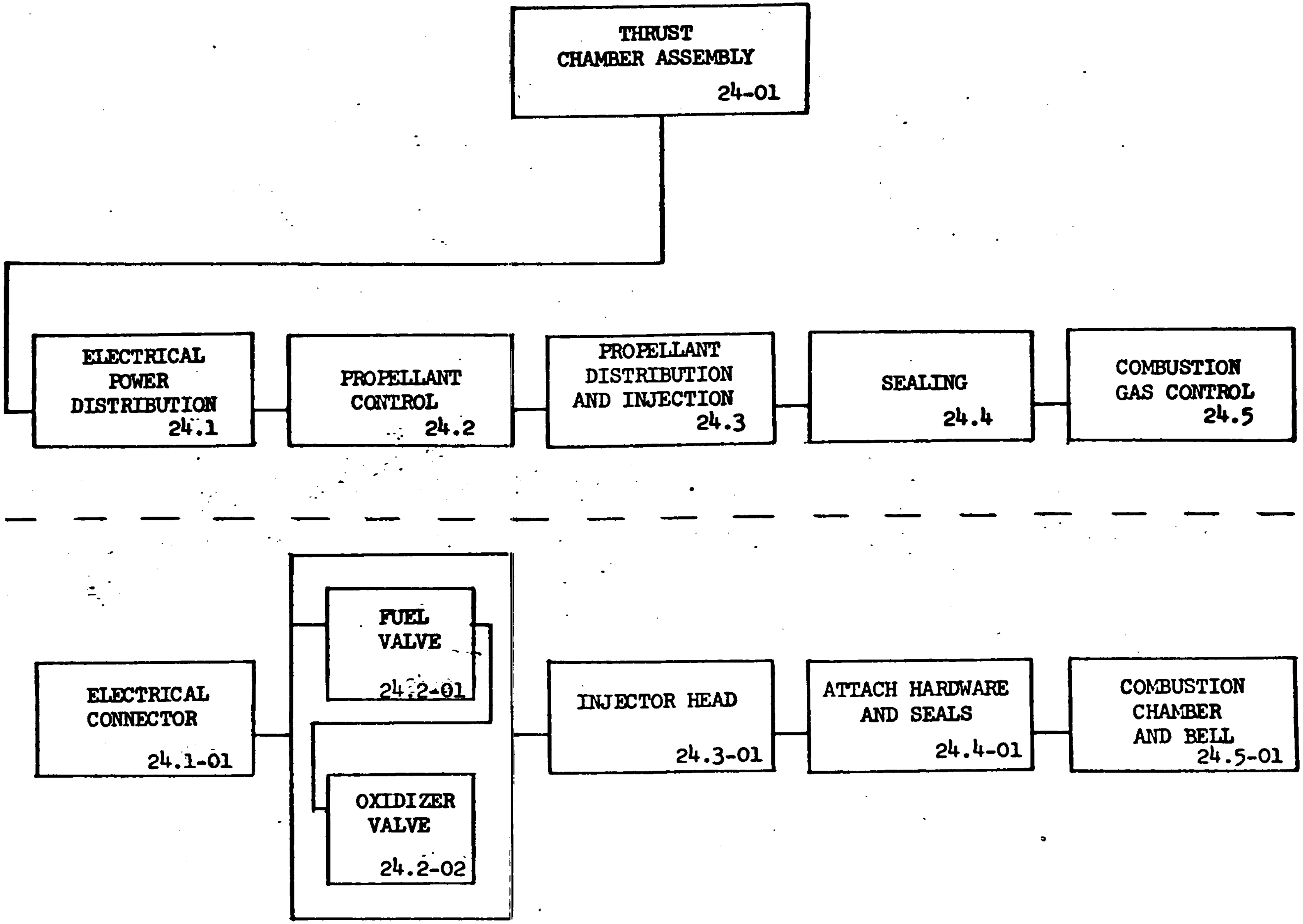


Figure 4.34. Reliability Logic Block Diagram Model R-4D Engine

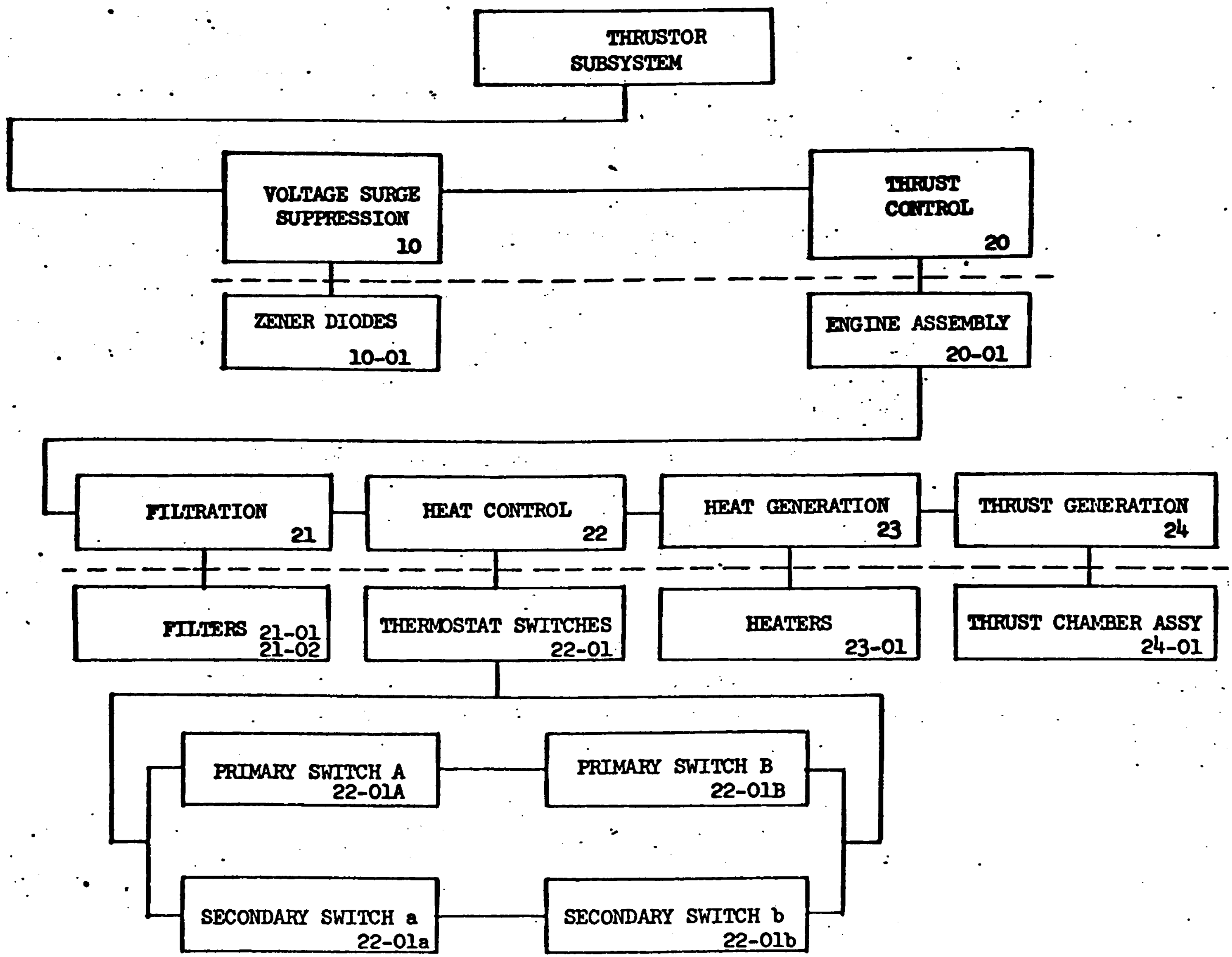


Figure 4.35. Reliability Logic Block Diagram Model R-4D Engine With Accessory Equipment

Table 4.22. Engine Reliability Estimates by Usage

Engine Duty	No. of Ignitions	Burn Time (seconds)	Reliability*
1. Spin-despin control	16	9,995	0.9999996
2. Precess	20	379	0.9999995
3. Stability control	10,059	1,000	0.99975

*The reliabilities shown above were based on tests totaling over two million engine ignitions, and 50 million valve cycles with no assessible failures.

much greater than the requirements for a given application (i.e., large margins) of safety so that engine reliability is a function of the required number of ignitions. The predominant failure mode is then valve failure to open or close. For the valve design, no assessible failures have been experienced in the extremely large number of tests encompassing over 2 million engine firings plus over 50 million valve cycles (Reference 4.3-1). Therefore, statistical methods were used to predict a minimum engine reliability. Engine reliability is presented in Figure 4.36, as a function of the required duty cycle or number of ignitions based on (1) the engine firing tests only, and (2) for engine firing plus valve cycle tests. Because of these presented arguments, the curve based on engine firings plus valve cycle tests should accurately predict engine-reliability. The unusual "bumps" result from use of six scale changes on the reliability abscissa; one change was made for each 9 units to enhance the accuracy in reading the reliability numbers. From the upper curve the engine reliability is taken as follows:

Required Duty Cycle or No. of Ignitions	Demonstrated or Achieved Engine Reliability (50 Percent Confidence)
1,000	0.999975
5,000	0.99988
10,000	0.99975
50,000	0.9988
100,000	0.9975

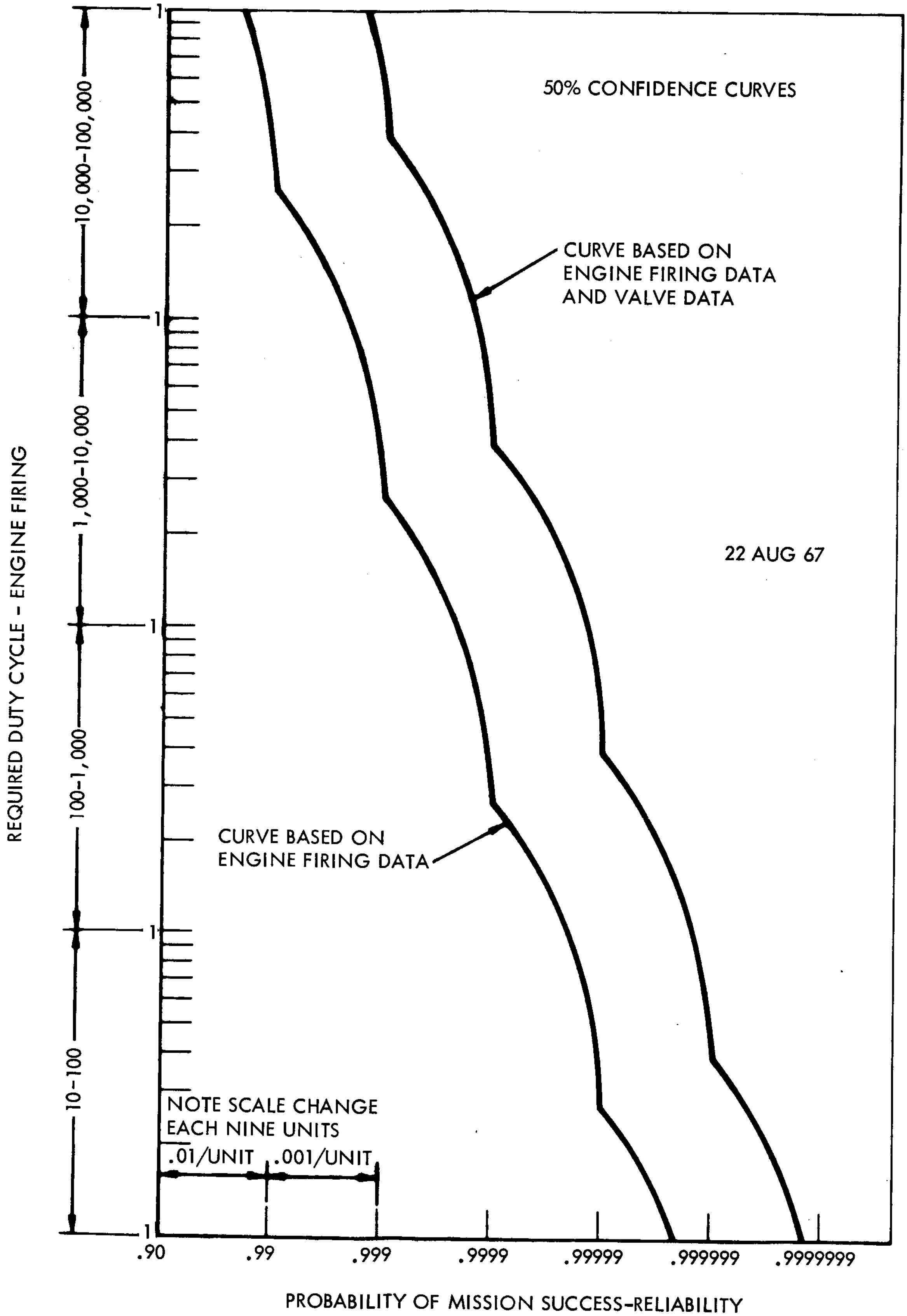


Figure 4.36. RCS Engine Mission Reliability vs. Duty Cycle

The engine exceeds the requirement by a substantial margin with resort to maintenance. The accuracy and extent of available data make it an attractive selection for any of the extended missions.

4.3.3 Engine Reliability/Life Factors

As previously indicated, three factors affect engine life and the resulting mission reliability and safety. These are burn time, space storage and the number of ignitions.

The predicted burn time limit of the molybdenum/molybdenum disilicide-coated combustion chamber, and the critical burn life limiting component of the engine is a function of maximum chamber temperature. The R-4D engine operates at a maximum chamber temperature of 2000 F. The predicted chamber life is in excess of 1500 hours; no application or margin test proposed to date would utilize even a small fraction of the engine's lifetime capability. This capability has been demonstrated by the supplier, see Reference 4.3-1.

The Number of Ignitions can also have a deleterious effect on engine life; this is normally created by the occasional rough start. However, since the beginning of the development program, there has never been a failure of any type in 1.5 million firings. This is not considered a failure hazard since the recorded number of successful firings is far in excess of the planetary mission requirements.

Deep space storage capability was one of the important design criteria for the engine. Materials and clamping loads in the hardware have all been selected to provide no degradation from stress or galvanic corrosion over long periods of storage, with or without engine firings. Space storage capability on the order of one year has been demonstrated on the Lunar Orbiter flights. During the past program, the engines demonstrated zero valve leakage in extended space exposure, as well as no ill effects due to the complete space environment (hard vacuum, micrometeorities, or solar radiation).

During long duration missions with intermittent engine operation and long off-times, it is possible to freeze propellants in the lines leading to the engines. This may be prevented if they are mounted in a plane with respect to the sun, so that all of the engines are warmed by the sun, or by intermittent engine operation every few hours. However, if the vehicle orientation is such that one side is in the shade and the engine is not used for long periods of time, the propellants may freeze in the lines. This may be avoided by using an electrical heater which is mounted on the engine injector head. Redundant thermostat switches are provided to regulate the head temperature between 65 F and 89 F. The heaters can be used as listed on the following page.

1. During long engine-off periods, the propellants can be allowed to freeze in the lines. The heater is then turned on prior to engine use to thaw the propellant.

2. The heater can be left on continuously, or

3. Thermostats can regulate the head temperature

There are no basic limitations on engine operations under any duty cycle except for propellant freezing as discussed above. The engines are unaffected by the presence of dissolved helium pressurant in the propellant. Variations in propellant supply pressure and voltages have only second-order effects on engine performance.

4.3.4 Engine Maintainability Potential

Although it has been shown that the engine displays sufficient reliability without M and R, it may be well to explore the possibility. The high reliability of the engine plus the redundancy obtained from having a number of engines capable of performing the same control functions reduces the need for replacement parts or repair of an engine during a mission. System reliability studies are expected to show that replacement parts should be assigned to other subsystems with lower reliability. To aid in such analyses, however, the in-flight maintenance possibilities based on the most likely failure modes requiring maintenance are considered.

1. Contamination in the propellant which is flushed through the engine could cause valve leakage from either blocking the valve in the open, or partially open position, or the contaminant could scratch the valve seat and leave a leak path. In the case of very slow leaks, the self-healing properties of the teflon seat will probably stop the leak before appreciable propellants are lost. If not, subsystem valves isolating the engine or engine cluster from the tanks could be closed during long periods of engine-off-time to reduce propellant loss. For larger leaks which could be detected visually, the engine should be removed and the valve replaced.

2. Micrometeorite hits or accidents from extravehicular activity or rendezvous maneuvers might damage the bell or molybdenum thrust chamber. Either of these parts could be replaced without removing the engine from the mount. However, this maintenance might be accomplished more conveniently in the vehicle workshop. In this case engine removal would be required. A large special-spanner torque wrench with a 3-inch jaw open and 200 ft-lbs. capability is required to remove and replace the bell. Only small wrenches with low torque requirements are required to remove the engine

from the mount, the thrust chamber from the head, or the valves from the head (Figure 4.37).

On the first four Lunar Orbiter flights using this engine, through 25 July 1967, 11 micrometeorite hits had been recorded by the detectors. Estimates of micrometeorite hit probability have indicated that the four engines should have been hit about 6 times; one or two of the hits should have been on the molybdenum chamber and the remainder on the bell. Since the engines have been fired recently and successfully, it would appear the engine can tolerate micrometeorites which are at least large enough to puncture the 0.001-inch-thick beryllium copper skin of the micrometeorite detectors. (Reference 4.3-1.)

3. Heaters and Thermostats - if operation is required for long time periods measured in months, it would degrade the reliability of the engine assembly and failures could be anticipated. These components are screwed to the injector head and could be replaced easily. However, missions and engine operations could be accomplished even with thermostat and heater failures by limiting the duty cycle or vehicle orientation.

Removal of the engine from its mount will require an accessible mount design, removal of four to six studs with the head ends pointing in either the fore or aft directions, unplugging the electrical lines, and removal of the threaded Dynatube fittings from the valves. The latter operation requires a special wrench of about 40-ft-lbs capability which can anchor against the valves to avoid attach screw damage. Prior to opening the propellant fittings on the valves, the cluster isolation valves should be closed and the engine valves opened for a minimum of one hour, to provide hard vacuum purge of propellants remaining in the lines attached to the engine. (See Figure 4.37.)

4.4 THE UP-DATA LINK SYSTEM

Note: Much of the information contained herein was provided by the Motorola Corporation through Reference 4.7.

4.4.1 Mission Functional Requirements

The primary function of the Up Data Link (UDL) is to provide the data required to update the Guidance Computer as it is received via the earth-spacecraft link. This is particularly important since the recommended operational mode involves shutting down the guidance and navigation system

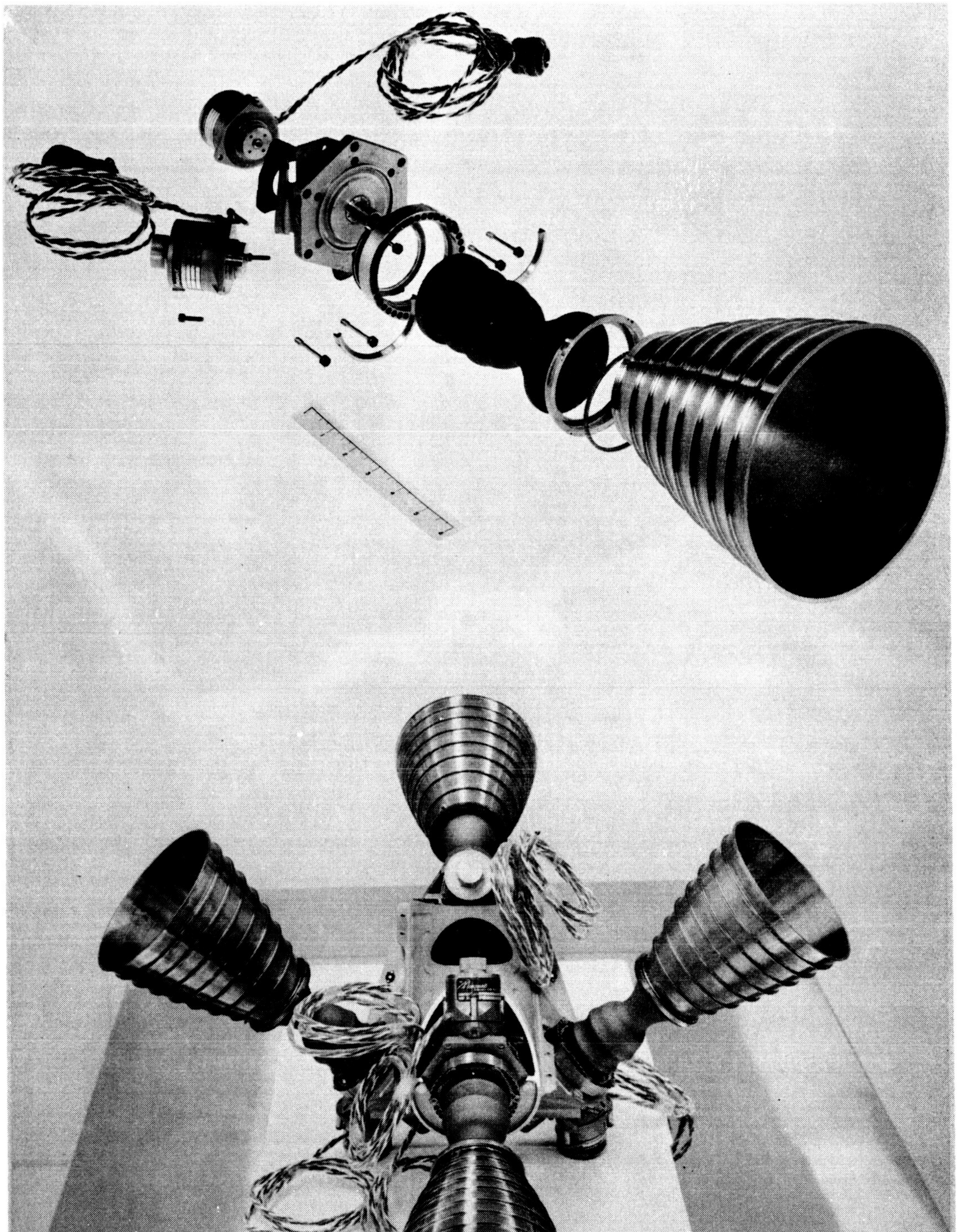


Figure 4.37. Rocket Engine Assembly, Exploded View

for a major part of the mission. This places primary responsibility for the navigation function on the earth tracking and computing complex, plus the earth to spacecraft communication link.

Specifically, the UDL performs spacecraft command and control functions by receiving the signal from the communications link, processing the data, and transmitting the commands to the spacecraft functions serviced. The types of commands to be handled are expected to be much the same as those used for Apollo, they will be coded by subsystem and type of command or data and will include the following.

1. Real time commands to control the operational mode of some spacecraft subsystems functions.

2. Insert words into the guidance computer to update and/or modify stored information. Words consist of a storage address, and either an instruction or numerical data; includes updated trajectory.

3. Update computer data.

The data provided by the earth complex depends on the Deep Space Net Tracking which is to be the primary mode of navigation for most of the mission. This function is particularly critical for the earth approach phase where the on-board major navigation capabilities are expected to be marginal. The projected functional requirements are given in Figure 4.38.

4.4.2 Mission Reliability Assessment

The reliability assessment is based on the assessed reliability of the Apollo system as reflected in Reference 4.8 and applied to the 700 day baseline planetary mission. The expected duty cycle for the UDL in the baseline mission is a 530.16 hours, maximum. Of this time, 0.56 hours is expected to involve a more severe environment resulting in a factor of 10 higher failure hazard for that mission phase.

The data used herein has been correlated with several other space programs involving similar digital command equipments that utilizes cordwood construction modules, using similar or identical type high reliability discrete components that have preceded the present Apollo Up Data Link design. Among these are:

Apollo Block I UDL (NAA P.O. M4J3XAX-406044).

Gemini Digital Command System (McDonnell Aircraft Corp. P.O Y20223-R).

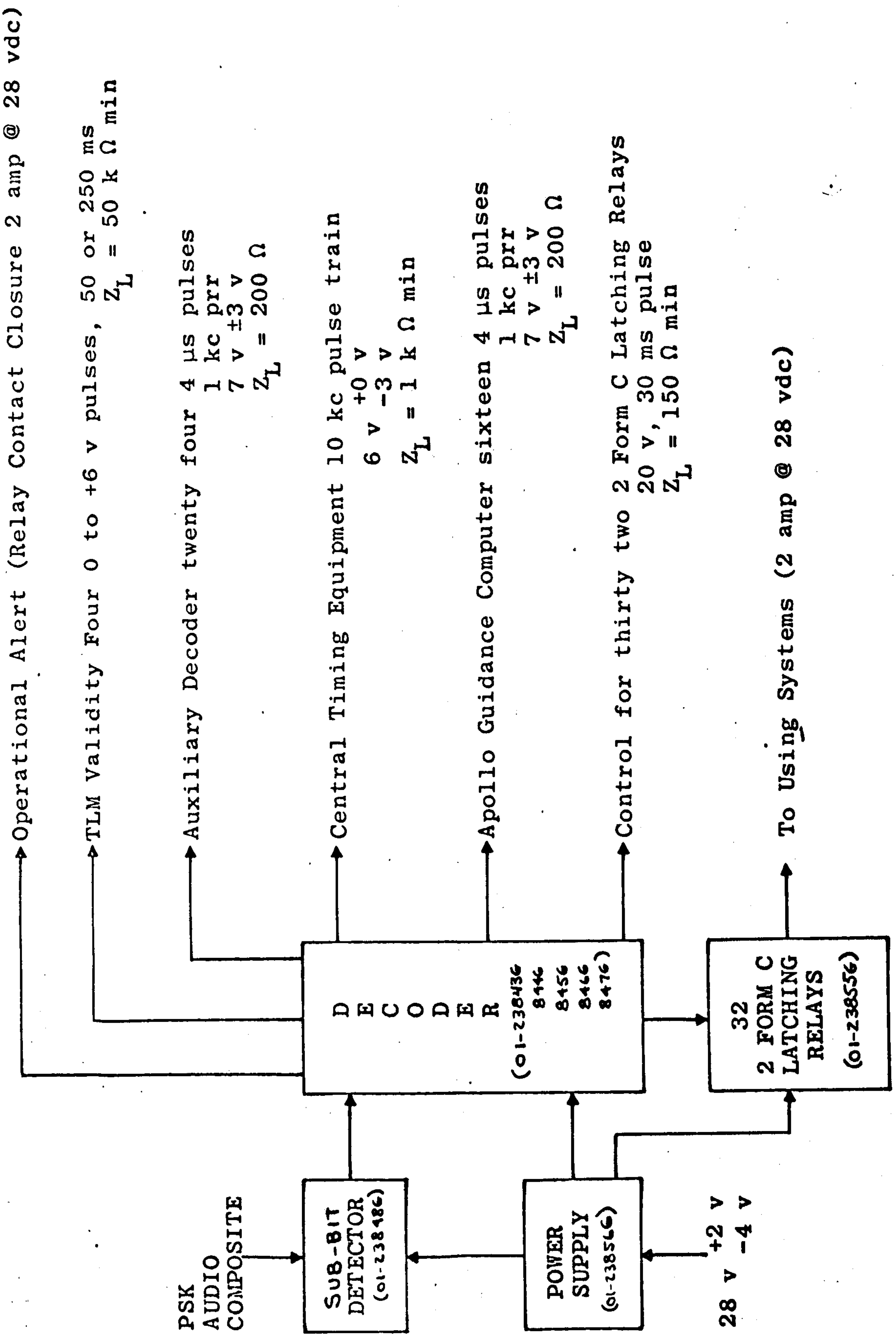


Figure 4.38. Up-Data Link, Functional Requirements

Agena UHF Command Receiver Type IX (LMSC P. O. 28-2539).

Mariner-Venus 1967 (Jet Propulsion Lab., Contract 370991).

Mariner C (Jet Propulsion Lab., Contract 950481).

Mariner R (Jet Propulsion Lab., Contract 950078).

Operational data has been acquired on components and assembly techniques throughout the phases of these programs to establish reliability history worthy of use in an extended mission. These data got to assure the validity of this analysis.

The resulting reliability of a UDL in the Apollo configuration may be computed using a failure rate of 1.076326 percent per 1000 hours, as defined for the unit. The probability of no failure during the mission is therefore 0.994.

Since the failure rate used represents an elevated temperature reliability number and the expected equipment temperature during the Mars mission will be limited to a normal maximum of 70 F, a new pessimistic failure rate may be achieved by applying a multiplier of 0.5 to the existing high temperature failure rate. It is likely, then, that the resulting reliability will be at least 0.997.

A breakdown of the system to subassembly levels is shown in Figure 4.39. Respective failure rates are shown in each block. Although the failure rates represent elevated temperature conditions, relative magnitudes may be easily determined to establish subassemblies that have a higher probability of failure than others.

The results of the reliability analysis indicates that the reliability of the UDL without maintenance or repair is nearly high enough to satisfy the mission requirements. That is, there is less than three chances in one thousand that a failure will occur to prevent accomplishment of the intended functions.

4.4.3 Availability Analysis

Despite the fact that the UDL will come close to meeting reliability and safety objectives without M and R, it seems desirable to consider the possibilities and the resulting influences on these objectives. To increase the potential reliability and safety of the UDL functions, several alternatives are possible, any of which will surpass the baseline mission objective by a substantial margin.

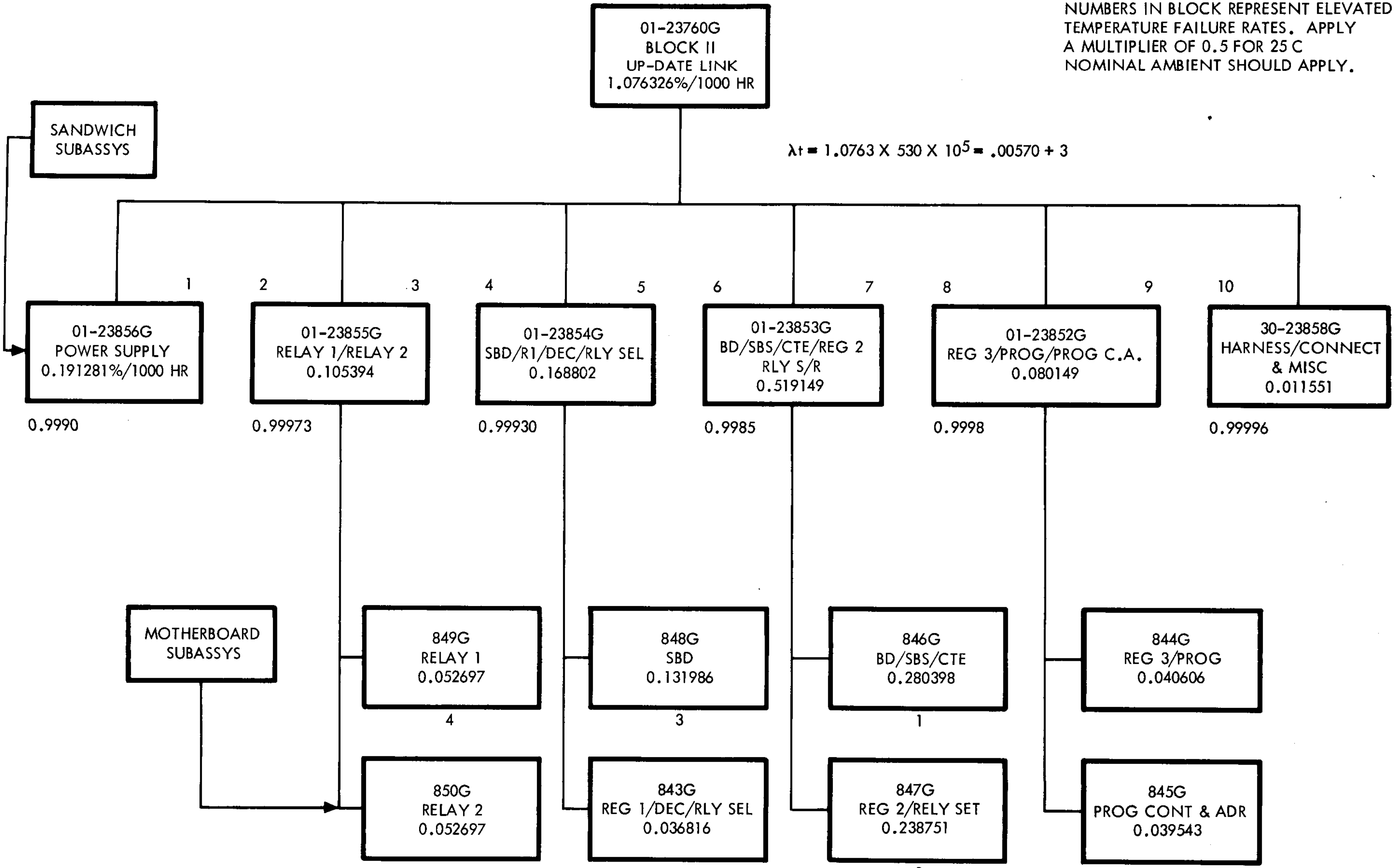
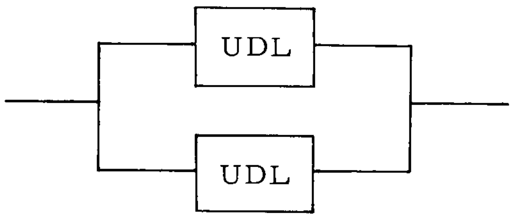


Figure 4.39. Reliability Tree, Up-Data Link, Mars Flyby Mission

1. Two UDL's operating in redundant fashion with paralleled output capability yields the resulting reliability (Rp_O) et 70 F: Rp_O = 0.9999917. This number represents a maximum estimated reliability since additional circuitry will be required in the UDL to allow outputs to be connected in redundant fashion. Therefore, this is a less desirable mode of operation from the reliability point of view, but somewhat safer because no information is lost in the switchover or replacement process, in case of failure.

2. Operate two UDL's in a standby redundant mode with paralleled output capability and manual or automatic switching between the two; or, as a third alternative, replacement of the total UDL package. Either of these will increase the resulting reliability and safety or P_s to:

   at 70 F: P_s = 0.9999919.



   Further considerations of redundancy is considered unnecessary since one spare or standby redundant UDL reduces the chance of loss of function to nearly eight in one million (8×10^{-6}).

3. Maintenance and/or Repair of the UDL can be accomplished at two levels in its present form, with minor modifications, as noted in a subsequent paragraph. Referring to Figure 4.39, it is evident that the two levels, in addition to the total component, are the sandwich level and the "motherboard assemblies." Also see two photos, Figures 4.40 and 4.41.

From Figure 4.23 it is evident that one sandwich assembly and/or three of the motherboards contribute over 70 percent of the mission failure hazard. The sparing concept could be established to work at either or both of these levels. From Table 4.28, it is evident that three of the sandwich boards must be spared to reduce the risk of no spare (P_S), to less than one in a thousand, and four motherboards were required to accomplish the same purpose. This is because the motherboard is rated at a lower level of assembly than the sandwich level. Of the two levels, the sandwich level may be the best selection, since one must be spared in either concept and, in addition, the diagnostic requirements are far less.

The total hazard could be further reduced to a level of insignificance, less than 8×10^{-6}, by providing 6 spare sandwich assemblies or 8 spare

Table 4.23 The UDL Sparing Analysis

Assembly	No. Req	Resultant UDL P_s	Weight (pounds)	Volume (cubic inches)
Basic UDL	1	0.994	19	600
Sandwich level				
01 - 23853G	1	0.997	2	80
01 - 23856G	1	0.998	2	80
01 - 23854G	1	0.999	2	80
Totals	3	0.999	6	240
Motherboard level				
B 46G	1	0.996	1	30
B 47G	1	0.997	1	30
B 48G	1	0.998	1	30
*01 - 23854G	1	0.999	2	30
Totals	4	0.999	5	120
All motherboards	10*	0.999998	10	400*
All sandwiches	6	0.999998	8	480

*Includes sandwich level where no further breakdown is practical - (two only)

motherboards with 2 spare sandwich assemblies, one for each in the UDL. However, this is unnecessary.

4.4.4 Maintainability Considerations

The UDL in its present form is maintainable to a limited degree. This may be seen from Figure 4.40 and 4.41. Minor modifications can improve its maintainability substantially. In the previous section, it was noted that only one sandwich board need be replaced and/or up to three specific

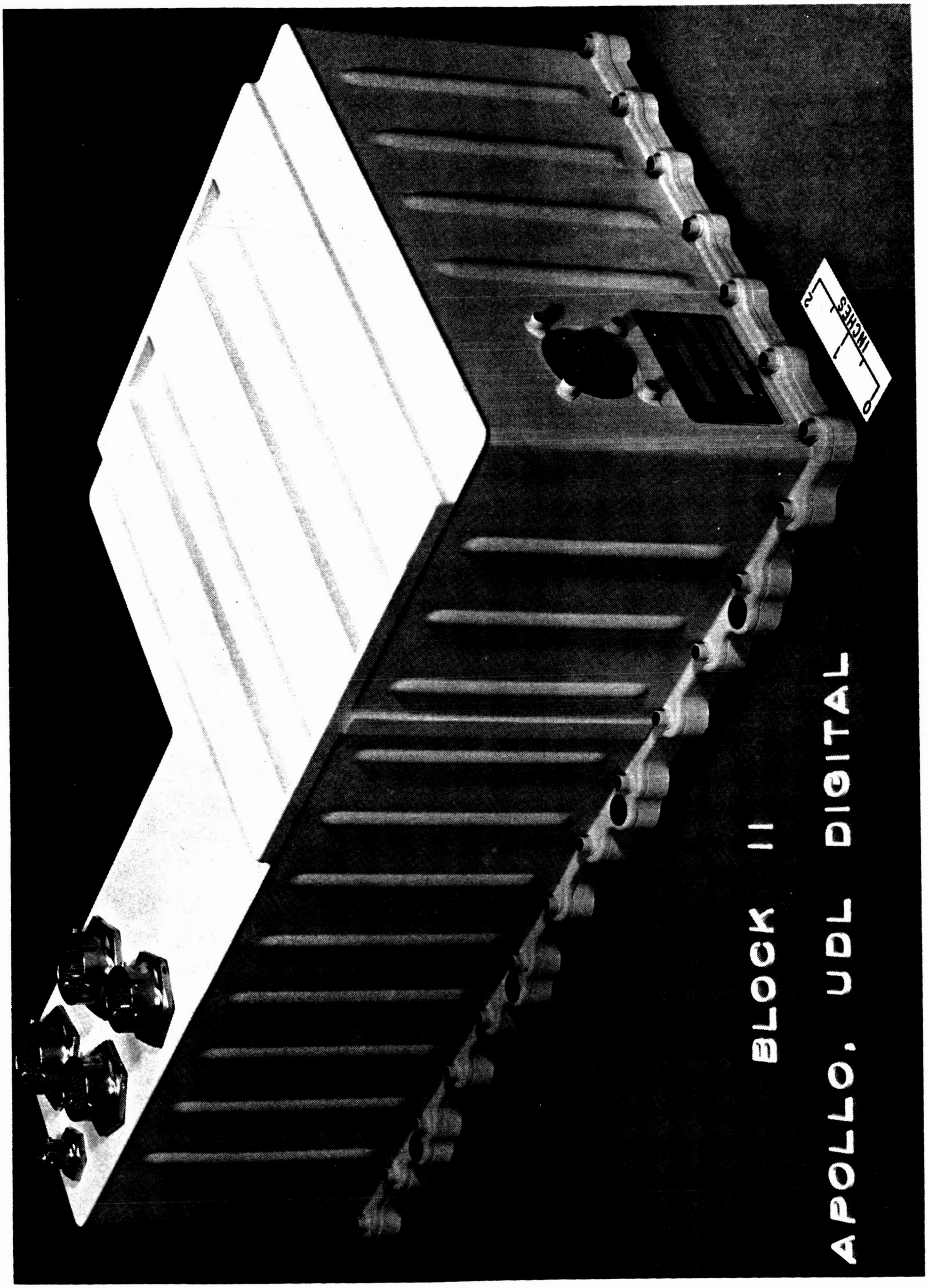


Figure 4.40. Photo, Apollo Up-Data Link, External View

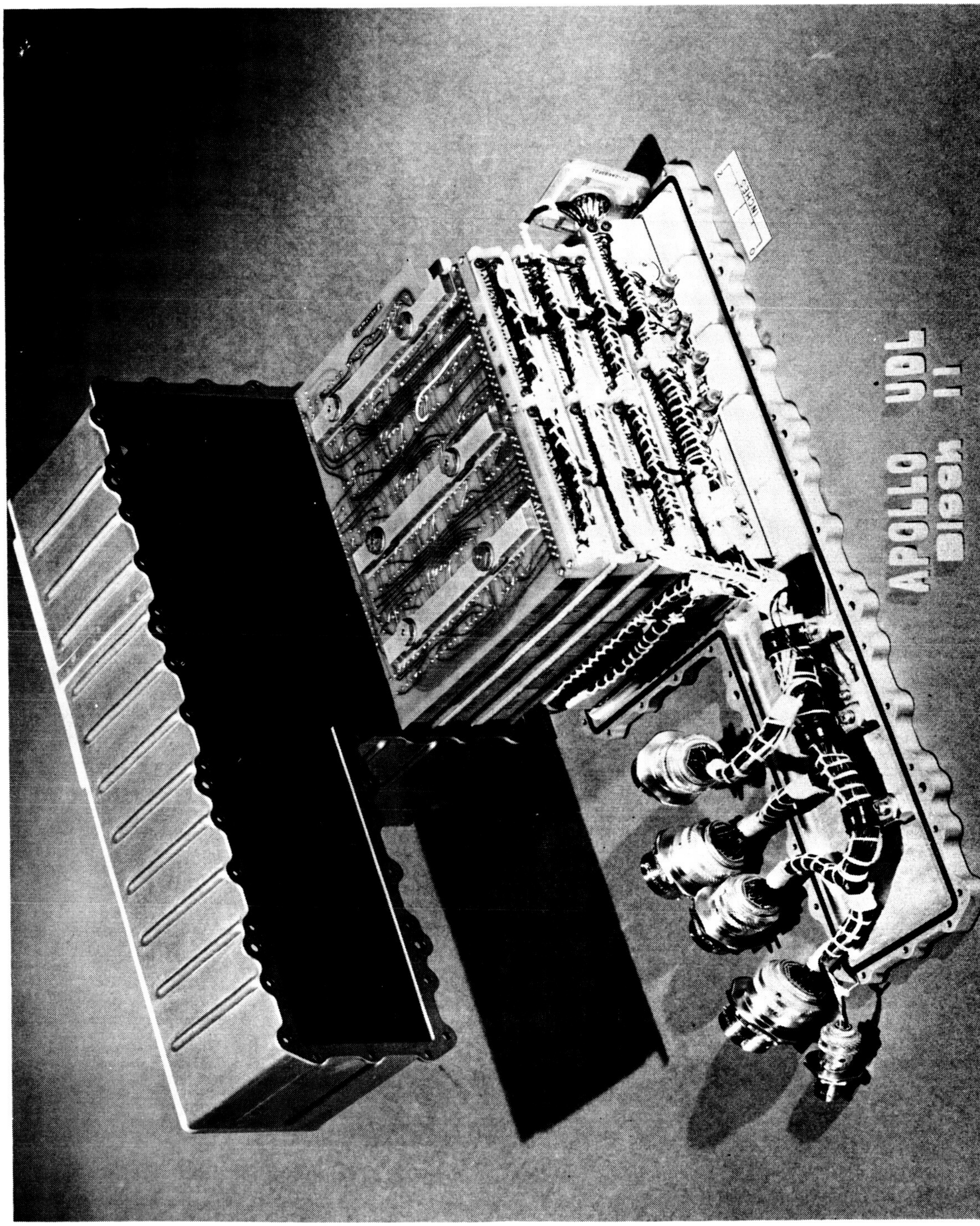


Figure 4.41. Photo, Apollo Up-Data Link, Open for Maintenance

motherboards which contained most of the failure hazard. Therefore, the following assumptions govern the maintainability requirements:

1. Reliability necessitates in-flight maintenance.

2. Retaining as far as possible the present UDL design of potted modularized discrete components mounted on printed wiring boards.

3. The lowest replaceable units required are the "sandwich" assemblies.

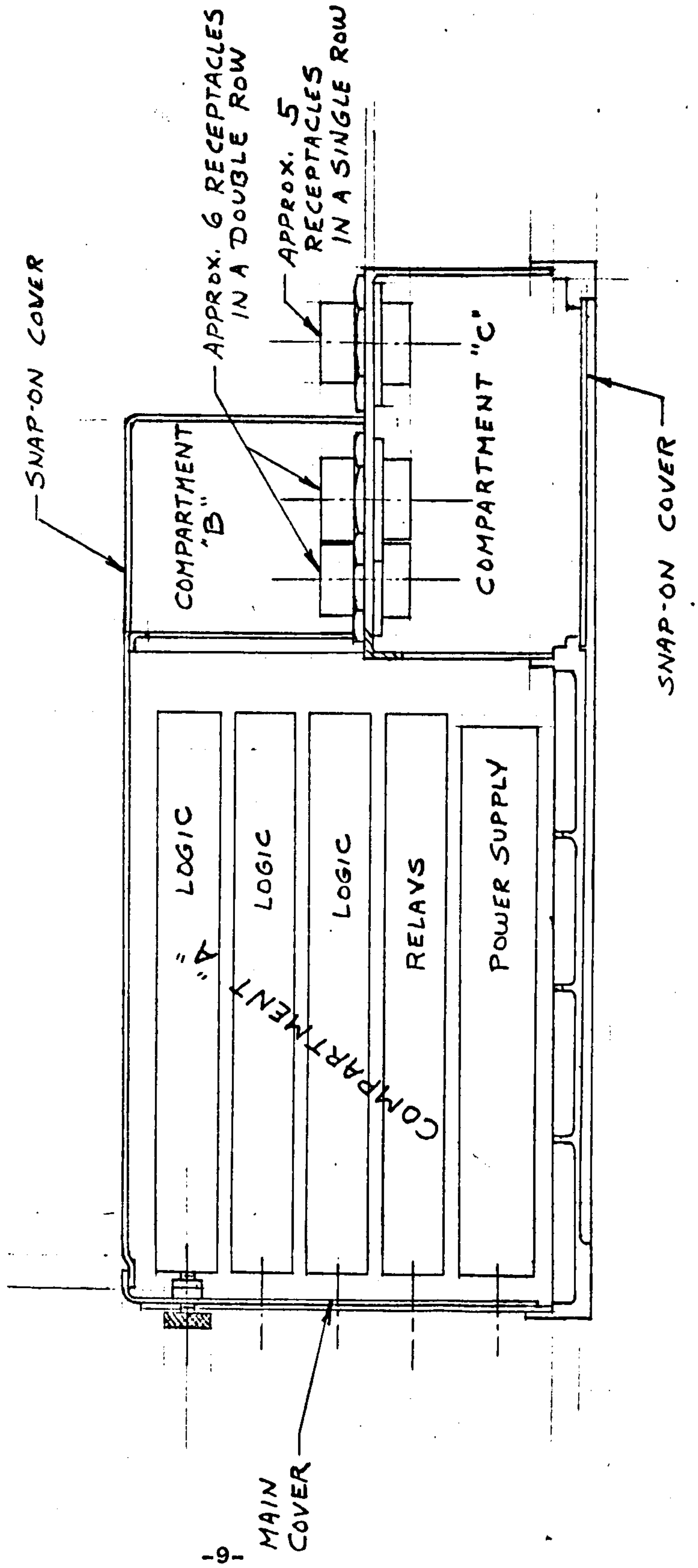
4. Environmental conditions do not require a sealed UDL housing, i.e., in the sense that it must be soldered.

5. A minimum usage of in-flight tools.

The present printed wiring board tracking layout and potted module design will be retained. Physically, however, the boards will be designed to mount in aluminum frames for mechanical stability, and also provide the means of locating the sandwiches on guide tracks in the housing. Stranded hook-up wire, hard-wired to the printing wiring boards will provide the external connections.

The sandwiches will be installed through the main opening at the left in Figure 4.42. Initial positioning will be provided by guide tracks on the side walls of the housing. The left to right positioning of the sandwiches will be by a wedge action provided by matching tapered surfaces between the guide tracks and the sandwich frames. These surfaces will be located at the end of the sandwich frames away from the main opening. The main cover will be installed by dropping it vertically in guides provided by the side walls of the housing. After the cover is slid down into position, knurled headed screws will contact the sandwich frames and force them against the tapered wedge surfaces at the opposite end. The sandwiches will thus be firmly secured against vibration and shock.

Each sandwich will have a harness lead assembly hard-wired to the sandwich printed wiring boards. The end of the lead assembly will provide an electrical plug for attachment to the appropriate housing mounted receptacle. The harness leads of the three logic sandwiches and the power supply will pass through the opening between compartment A and compartment B and be connected to the top side of the feed-through receptacles between compartment B and C. All required interconnections between sandwiches will be done in the harness assembly in compartment C on the lower side of the internal receptacles. This harness assembly also provides the connections to the external connectors, shown in a single row to the right. The relay subassembly will have a branched harness lead assembly, part passing



SCALE: APPROX. HALF SIZE

Figure 4.42. Wiring Board Description

into compartment B and all outputs going directly through the opening into compartment C to the external connectors. All the plugs will have knurled rings for bayonet locking.

Only approximate estimates of size and weight have been made. The dimensions are 14 inches long by 10.5 inches wide by 6.5 inches high. The volume is 900 cubic inches. The total new weight for the modified UDL is estimated at 35 pounds.

The UDL assembly, when repackaged for the recommended maintenance, will have only snap-on covers and a main cover that slides into place, locking itself and the sandwich subassemblies by means of the knurled finger screws. All connector plugs will be attached by hand. Therefore, no tools are required for disassembly and reassembly for in-flight maintenance. The only wrench required may be a standardized tool necessary for removing the UDL base mounting screws if the unit is in a location in the vehicle that makes disassembly in place impractical. However, this is not a severe requirement since a single standardized tool could be designed for removal of all equipments in the vehicle.

4.4.5 Conclusions and Recommendations for the UDL

The data presented herein demonstrates that the Apollo UDL will perform satisfactorily on the longer missions. For the 700-day baseline mission, a modest sparing concept will virtually eliminate the hazard. Maintenance is possible but some repackaging is recommended to facilitate maintenance at the sandwich level. The following recommendations are suggested to facilitate maximum availability of the UDL:

1. The UDL should be repackaged into a form factor previously recommended - thus providing for faster removable subassemblies

2. Since the UDL has no power-on "idle" mode, it is necessary that power be applied only when the unit will be operated so that the integrity of the reliability goal is to be preserved. The number of power on/power off cycles is low and should not create any reliability degradation.

Table 4.24 describes the different use considerations for the UDL, and respective estimated parameters. Although it was intended that the electrical design should not incur any changes, mention is made of the integrated circuit designs to illustrate the impact of size and weight reductions, should these savings ever be required.

Table 4.24 Patented UDL Concepts and Associated Applications Considerations

Configuration	Weight (lbs)	Volume (cubic inches)	R at 530 Hrs	R at 700 Days
Single Block II UDL (as is)	19	600	0.997120	0.8340
Dual Redundant Block II UDL (operating)	44	1640	0.9999917	0.9724
Dual Redundant Block II UDL (standby)	38	1200	0.9999919	0.9848
Single Block II UDL (repackaged)	35	900	0.997120	0.8340
Dual Redundant Block II UDL (repackaged, standby)	70	1800	0.9999919	0.9848
I/C Block II UDL (repackaged*)	12	260	0.9971	0.8340
Dual Redundant I/C Block II UDL (repackaged, standby)	24	520	0.9999919	0.9848

*I/C UDL developed under NASA/Houston contracts NAS9-3468 and NAS9-5366

4.5 PROPELLANT MANAGEMENT SYSTEM

4.5.1 Mission Functional Requirements

The propellant management system involves all of the functions required to store, control, measure and disperse all propellants. Propellants must be provided to three basic systems, the reaction control engines, the gravity control engine, and the main propulsion engine. Each function operates under a different set of circumstances and therefore impose different requirements and constraints on the propellant management system.

The Reaction Control Engine must work in the zero-gravity mode which imposes the need for a positive expulsion feed concept for a total of about 900 hours. This is divided into five periods, the longest of which is 460 hours.

The Gravity Control Engines can operate under one of two modes. Either a zero-gravity condition can be assumed, as will be the initial state, or, it can be planned to use the RCS engines to provide ullage control and then operate as if under a gravitational field. This is feasible and perhaps the most desirable since the spinup/despin time will take about 1.2 hours per cycle. For purposes of this study, the ullage control mode will be assumed.

The Main Propulsion Engine(s) can be operated under the same conditions as the Gravity Control Engines and, since the Apollo concept involves ullage control and uses it successfully, it seems desirable to apply the same concept to the longer planetary missions. No change in conditions are expected to influence this situation. Therefore, the propellant feed system associated with the Apollo Service Module will be applied to the baseline mission herein.

4.5.2 Positive Expulsion Tankage

Note: Much of the data used herein was provided by the Bell Aerosystems Company of Buffalo, N.Y. through Reference 4.9.

1. Functional Usages

   A study of positive expulsion tanks is presented to provide information on the feasibility of using and/or modifying the Apollo systems from the present configuration to a 700-day mission duration capability.

As presently configured the positive expulsion tank (bladder type) is loaded and pressurized (suppression pressure) prior to launch. Depending upon tank application the pressure may be increased to its operating level either prior to or subsequent to launch. From this point the pressure remains constant enabling positive propellant expulsion on demand. Pulse type flow is the normal requirement and is effected by operating a valve in the reaction control thrust chamber subsystem. Apollo mission requirements include the tank assembly capability to perform satisfactorily for one cycle, i.e., a propellant loading ullage draining and complete expulsion to full bladder ΔP. Figure 4.43 is a functional block diagram of a typical positive expulsion tankage subsystem.

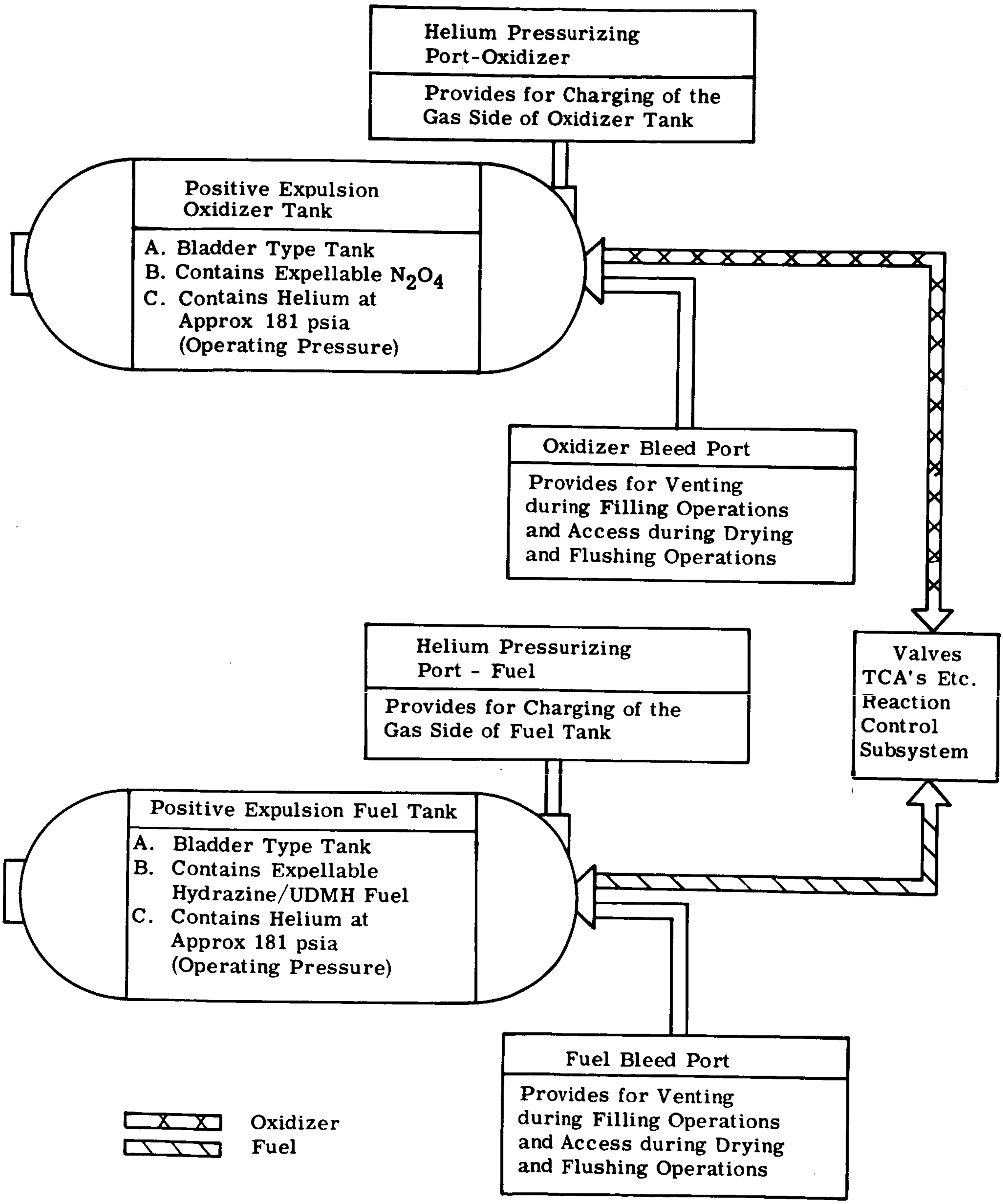
Whereas the Apollo spacecraft contains six different (slightly) configurations of the positive expulsion tank concept, their basic design is the same. Therefore, for the sake of clarity, one specific design is presented (Figure 4.44) to enable proper identification of parts and correlation with remarks in the body, table and figures in this analysis. Pertinent difference will be identified where applicable throughout the report.

The operational usage for the baseline long duration mission is expected to vary from the Apollo usage. For missions employing the artificial gravity modes the RCS is only used during zero gravity flight, the system can therefore, be depressurized for the major part of the mission. This is expected to relieve the reliability problem since pressurization with Helium creates a major failure mode by migrating into the propellant. Pressurized operation is required for no more than 900 hours out of the over 16000-hour mission. A functional diagram is presented in Figure 4.45.

2. Reliability Assessment

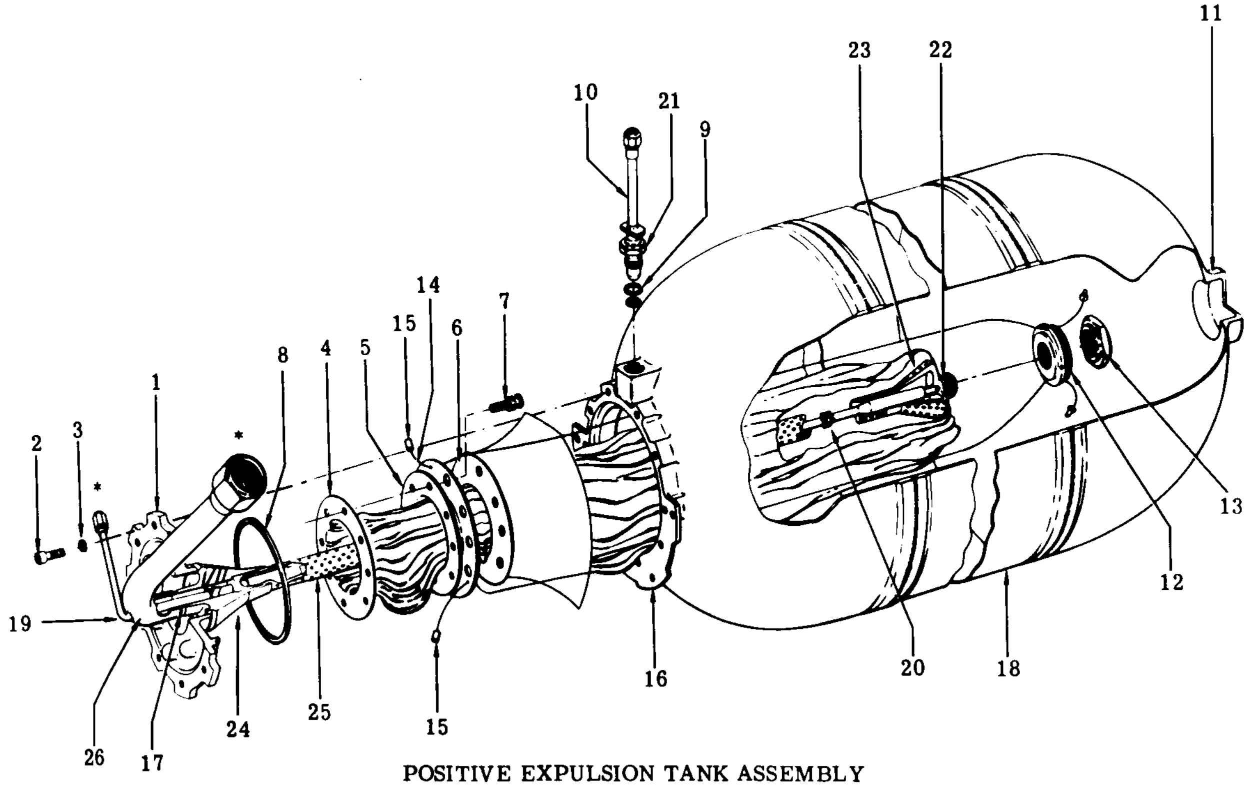
Using the Apollo spacecraft system as the baseline, the reliability logic (block) diagram was established and appears as Figure 4.45. From the diagram, note that the tank assembly has been divided into three main subassemblies (tank, bladder and diffuser assembly) with their associated failure rates. Subdivisions are listed to correlate with the failure mode and effects analysis. The diagram is a simple series arrangement since no redundancy exists for the primary function of propellant expulsion. The two vent lines are a servicing aid only.

The failure rates for the tank and diffuser assembly fallout is insignificant in arriving at the tank assembly reliability. Potential problems peculiar to extended missions are discussed in subsequent portions of this report. Therefore, tank assembly reliability is primarily a function of the bladder failure rate and it has been demonstrated that, given a satisfactory leakage check and propellant loading, a successful full expulsion (pulsed or continuous) can be effected with a 0.9976 reliability at 90-percent LCL.



POSITIVE EXPULSION TANKAGE FUNCTIONAL BLOCK DIAGRAM

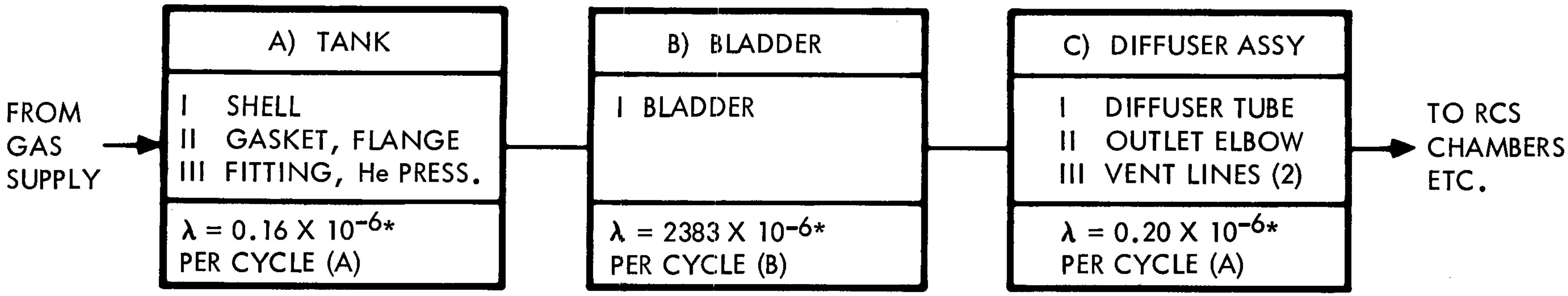
Figure 4.43. Positive Expulsion Tankage Functional Block Diagram



POSITIVE EXPULSION TANK ASSEMBLY

1. Diffuser Assy.
   Fuel 8339-471053-3 *nut cut off
   Ox. 8339-471054-3 predelivery
2. Bolt 8271-471019-1
3. Washer AN 960-C416L
4. Bladder
   Fuel 8339-471080-1
   Ox. 8339-471080-3
5. Ring 8271-471117-1
6. Pad 8271-471249-1
7. Bolt 8271-471114-1
8. Gasket 8271-471025-1
9. Gasket 4G2-4 (Test Only)
10. Gas Inlet Fitting (Test Only) 8339-471026-1
11. Boss End of Tank Assy.
12. Washer Assy. 8271-471144-1
13. Nut 8271-471021-3
14. Vent Line Assy.
    Fuel 8339-471027-1
    Ox. 8339-471027-3
15. Eyelet 8271-471141-1
16. Mounting Flange
17. Bimetal Joint (6061 Al.Al. to 347 St.St.)
18. Tank Shell Assy. (T16AL-4V Titanium)
    Fuel 8339-471110-1
    Ox. 8339-471110-3
19. Bleed Tube Assy.
    Fuel 8339-471059-9 * nut cut off
    Ox. 8339-471059-11 prior to delivery
20. Bleed Tube Spacer
    8339-471058-1
21. Nut 8271-471049-1 (Test Only)
22. Retainer End of Diffuser Assy.
    (Ret. Assy. 8339-471057-1)
23. Retainer Cone 8339-471057-5
24. "Outlet End" Cone 8339-471034-1
25. Diffuser Tube 8339-471036 (-1 Fuel; -3 Ox)
26. Elbow (Outlet) Tube W/O Nut & Sleeve 8339-471031-7

Figure 4.44. Positive Expulsion Tank Assembly



TANK ASSEMBLY MISSION FAILURE RATE:

$\lambda = A + B + C$

$= 2384 \times 10^{-6}$/PER CYCLE AT 90% LCL

TANK ASSEMBLY MISSION RELIABILITY

R = 0.9976 AT 90% LCL ASSUMED ONE CYCLE

*CALCULATED FROM BAC DATA: 965 PROPELLANT EXPULSIONS WITH ZERO RELIABILITY FAILURES UTILIZING 41 TANK ASSEMBLIES (MODELS 8271, 8339 AND 8400).

Figure 4.45. Reliability Logic Block Diagram, Positive Expulsion Tank

It is noted that this data was not derived from the one specific design referred to earlier but rather is based on five of the six configurations in the spacecraft plus two from the launch vehicle. The one from the spacecraft not considered (CMO) contains a bladder with 9-mil-thick hemispherical ends with a 6-mil cylindrical center section, as opposed to the others which are "uniformly" 6 mils in thickness. The two launch vehicle tank assemblies were included since their bladders are identical (same part number) to that shown in Figure 4.44. Calculations are based on 965 actual propellant expulsions (fuel-MMH or 50/50 blend; oxidizer - N_2O_4) with zero reliability failures. The 41 test tank assemblies employed were expelled under various combinations of attitude, temperature and prior dynamic testing.

A failure mode and effects analysis - FMEA was conducted on the tank function components to identify the potential and its "operational effect" on the equipment and system and on the Mission. The problem was isolated to the mission lowest affected level. In all cases, except one, where a mission effect exists at all, the lowest affected level of assembly is the tank assembly, in toto. The one exception is a crack in the diffuser tube (without a separation of the parts); however, it should be noted that with respect to repair, if desired, the entire assembly would be affected.

The analysis is based upon an occurrence of a single failure mode. By anticipating all of the possible failure modes and providing inherent compensation (adequate design margins) and checkout procedures, the occurrence of failures that will jeopardize the successful completion of the baseline mission is eliminated.

3. Availability Analysis

As noted earlier, the assembly reliability is a function of the bladder failure rate since the probability of tank or diffuser assembly failures is essentially zero. Improving availability, therefore, centers around the bladder, with the potential problem of long term permeation (bubble formation) as the area of most concern.

For the sake of the analysis it is assumed that the bladder reliability is not adequate for the mission. This approach results only from the standpoint that long-term storage/usage has not as yet been demonstrated.

Maintaining the assembly at or near its operating pressure has been shown to postpone effectively the formation of a gas bubble resulting from helium permeation to the liquid side. Since the length of postponement is not presently certain and since there is no method to detect this short of using an X ray, it appears that periodic timed re-bleeding during non-operating artificial g (1/3-g) phases is the best solution, and will resolve the problem. Whereas this would not require accessibility, if a remote control vent valve were employed on the bleed tube, it would necessitate proper tank orientation (installation) and a sufficient helium supply to compensate for losses.

Two methods of improving the availability and safety of the positive expulsion tanks are possible. The first involves replacing the tanks as an entity when failed. This would involve storage and handling of tanks which are sensitive to mishandling. The second involves use of redundant design with a crossover feed systems. The first would increase the safety and reliability of the system to greater than 0.9999976 if the handling hazard is neglected and the redundant designs would be 0.9999942. Although the redundant design displays almost twice the failure hazard of the maintainable design, it is more desirable because of maintenance problems; see subsequent section. Further, the risk of the non-functioning RCS is less than 6×10^{-6} and since the proposed design includes two for each axis, the probability of no failure for each axis is for all practical purposes 1.0 (3.6×10^{-12}).

4. Maintainability Considerations

Detailed maintenance and repair of these type of tanks is impractical. Bladder type tank assembling typically requires a two-man effort in an ambient controlled clean room environment. The thin wall shell construction, with a fragility likened perhaps to that of an egg shell, is susceptible to even minor mishandling dents or scratches. Propellant loading temperature is +40 to +100 F. Storage and shipping are restricted to -20 to +110 F and +20 to +120 F respectively, with the tanks empty and a slight internal pressure. These restrictions are primarily a result of bladder limitations (brittleness at cold temperatures for example), and "storage" conditions specifically limits the safe handling of the tank assembly. These characteristics, are not the best for in-flight maintenance or repair when conducted in a zero-g "contaminated" environment, below the level of the integrated tank assembly.

Although gas leakage at the tank flange gasket is not a primary concern, it does, however, serve to illustrate some general requirements pertinent to maintenance or repair activities. Given, (a) tank assembly accessibility, (b) monitoring on diagnostic equipment - a mass spectrometer, for example, and (c) proper tools - a torque wrench and wire cutters, the flange bolts could be retorqued (55 to 60 in-lbs). Similarly, proper plumbing and existing system pressure, or a by-pass/pump system, would allow transfer of propellant from a suspected (or proven) faulty tank to a spare assembly. The spare assembly might well be a previously exhausted unit. "Exhausted," in this case, refers to approximately 1/2 of the propellant remaining to assure proper bladder re-expansion and loading. Assuming a failed bladder could be detected (liquid leakage to the gas side), we must now consider the practicality of spare parts - bladders and flange gaskets, in this instance. Storage of gaskets should present no problem because their weight and size are "negligible." Bladders are normally stored inflated by means of an internal slightly pressurized balloon to prevent the possible permanent set of creases or folds which could result if collapsed for long periods. The use of a balloon also precludes undue bladder stress.

Since the volume of an inflated bladder is only slightly less than a complete tank assembly, the choice of sparing would seem to center around their relative weights - approximately 0.75 pounds (plus gasket and balloon) versus 10.34 pounds (tank assembly) for example. However, in the light of the maintainability limitations cited earlier, where assembly mishandling and bladder replacement can pose problems, a trade-off may well favor sparing the assembly. This is particularly advantageous since no handling would be required if its storage area were also its use area. That is to say, it could be plumbed in as part of the system installation, but maintained isolated and slightly pressurized. In this condition it would not only be free from the possible unknown effects of long-term bladder exposure to propellants but would also serve in place of the aforementioned "exhausted" receiver unit.

5. Design Considerations

Since there are no known failure modes which cannot be precluded or eliminated through the inherent compensating provisions, adequate checkout procedures or in-flight maintenance, no design changes appear necessary at this time.

In view of the unknown potential problems, several produce improvement considerations are offered. Regarding long-term bladder exposure to propellants and radiation, an added laminate of aluminum foil shows much promise. (This could also serve as a permeation barrier in both directions, if this proves to be a problem.) Metal bellows offer the same qualities and also possess reloading capabilities unaffected by assembly attitude or zero-g environment. This would introduce a substantial weight increase, but is worth considering if reloading is desired.

Deterioration effects on the titanium shell from sublimation could be reduced through increased wall thickness, a protective coating, or a material change. (Increased thickness or a coating also reduces susceptability to denting or scratching; materials such as aluminum or stainless steel also afford longer term protection against the possible effects of propellant contact from a damaged bladder.)

In considering any design changes or improvements, it would be well to assess the relative merits of standardization throughout such as (a) identical tank assemblies for both oxidizer and fuels, (b) identical fuels (the oxidizer is already common), (c) substitution of a welded joint for the flange gasket, (d) the avoidance of the use of dissimilar metals to preclude galvanic action considerations and, (e) use of proper orientation of all tanks from the artificial g and horizontal vacuum propellant loading standpoints.

Incompatibility of the tank shell with the oxidizer (N_2O_4) was formerly thought to be a problem. However, curtailment has been accomplished by negating the effect of propellant vapor permeation by either controlling the NO content or peening the shell. Use of a different oxidizer is a possible improvement which gives rise to some interesting speculation regarding results of past Bell test programs. In all cases where actual stress corrosion and tank rupture took place, the titanium shells had been exposed directly to the N_2O_4. This resulted from loading the N_2O_4 into shell assemblies which did not contain bladders.

Omission of the bladders was a test expediency and obviously preceded awareness of the stress corrosion problem. The facts that if the N_2O_4 itself permeates, or if only some of its constituents actually transfer, may be detrimental to shell life. To illustrate the point, investigations of fuel tank assemblies with bladders showed that MMH does not permeate. Rather only some of its constituents do so, notably CH_4 (methane) and NH_3 (ammonia). Additional testing would determine if N_2O_4 follows a similar pattern, and therefore, its suitability can be established.

6. Recommendations and Conclusions for Positive Expulsion Tankage

The results of this study indicate that the Apollo RCS tanks are capable of meeting the requirements of a 700-day mission without significant redesign and contribute no assessible hazard to crew safe return. This is based on the analysis of all known potential failure modes which indicates that they can be precluded through adequate design margin and through check-out procedures, or nullified by in-flight maintenance - provided that the recommended provisions can be incorporated.

For example, to eliminate the potential problems of flange gasket leakage accessibility for retorque is required. For gas bubble formation on the liquid side, it is necessary that the tanks be properly oriented with respect to the artificial-g mode and a remote control vent valve be added to the system.

An assessment of the extended mission also reveals several unknown areas relative to environmental effects, such as long-term bladder exposure to propellants and radiation. Therefore, further studies are required to establish firmly the useful operating lifetime of the present configuration for extended mission applications. These include such tests as:

1. Long term material/propellant compatibility, external gas leakage, galvanic activity, bladder permeability, and shell sublimation.

2. Tendency of excessive bladder twisting experienced in assemblies requiring "horizontal" propellant loading (by vacuum) and expulsion.

3. Effects of repeated dynamic loads (a "time" rather than an "intensity" consideration) imposed on the bladder.

4.5.3 Main Propellant Tankage (Storage)

Note: Some of the data used herein was provided by the Allison Division of General Motors through Reference 4.10.

1. Functional Usages

The main propellant tankage is required to store both the fuel and oxidize, in a liquid state, for the full mission duration. Final fuel depletion is to be accomplished during earth approach for retrograde, and as a means of reducing the magnitude of the reentry velocity. The fuel will be used for about four mid-course corrections, one at each periplanet, one at earth approach, and finally, retrograde. Since the engine is pressure fed, helium will be used as a pressurant to expel the liquids from the tanks, after ullage control. At least two are required, one for oxidizer and one for fuel; redundancy is expected to be required for each. For a schematic, see Section 4.5.5 on the Propellant Feed System.

The operational concept involves relief of the helium pressure during the long coast periods and activation just prior to igniting the engines. Ullage control is accomplished through use of the appropriate reaction control engines just prior to main engine ignition.

Tank volume could be designed to meet mission requirements by merely varying the length and/or number of the cylindrical tank sections welded to the hemispherical domes.

2. Reliability Assessment

The reliability study for the Apollo Tank program is summarized in the Allison Final Design Report, Reference 4.5-3. The reliability of the propellant tank is estimated to be 0.99999974 compared to the program reliability goal of 0.999995. The reliability estimate applies to the actual use phase of the tank life, that is a propellant tank which has passed all inspections and acceptance tests and has been checked out in the vehicle has a 0.99999974 probability of going through the preparation, the countdown, the earth launch, the transfer, the lunar orbit, and the lunar landing without malfunction.

The reliability estimate and failure mode-failure effect analysis indicated that the probability of failure is nil for all modes of failure, with the exception of the burst mode of the tank pressure wall. This has a failure probability of 0.262×10^{-6}. The failure probability of the tank wall was calculated by the method of interference, using the probabilistic distributions of the tank stress and material strength.

Except for the 700-day mission duration, the propellants and tankage environment are assumed to be essentially the same for the baseline and Apollo missions. Consequently, the Apollo tanks should be directly applicable to the planetary missions provided the recommended testing reveal no detrimental effects of the extended mission time.

The reliability estimates for storage tanks are always suspect because they are not amenable to calculation by the normally accepted methodology, i.e., $R \neq e^{-\lambda t}$ for storage tanks. Rather, for the longer mission in particular, this results in a very pessimistic estimate and the true value (all the other factors being considered) is always much higher. However, to establish some boundary values, that value has been calculated and found to be 0.9956 for the baseline mission. Therefore, the expected value is known to be somewhere between the 0.9956 and 0.9999997 and known to be near the latter since it was calculated on the basis of strength-stress relationships and can be controlled by changing wall thickness.

3. Availability Analysis

Maintenance is not required, nor applicable. Availability of the storage function has been improved by relief of the helium pressure when not required. This act should result in raising the potential safety to nearly the Apollo value because approximately the same number of burns are required. This depends only on the result of the recommended test.

4. Conclusion and Recommendations

Apollo tankage designs are satisfactory for the proposed planetary missions. The safety estimates are expected to exceed the requirements by a substantial margin. Redundancy is expected to be desirable since the one potential failure mode could be negated by not pressurizing a given tank until its fuel is required. One tank set would thereby not be used until the final phases of the mission.

Qualification tests and propellant storage tests have established the suitability of tanks for the Apollo Mission. To substantiate the suitability of this tankage for the longer duration Mars Flyby Mission it is recommended that the following testing be considered.

1. Long-term creep testing of titanium material specimens, including welded specimens.

2. Long-term stress corrosion testing of welded and nonwelded titanium specimens in each propellant.

3. Long-term storage of propellants in tanks at design stress. These tanks could be either full-scale tanks or sub-scale tanks made by the same fabrication methods as the Apollo tankage.

4.5.4 Propellant Gaging Function

Note: Much of the data used herein was provided by the Simmonds Precision Products, Inc. through Reference 4.11.

1. Functional Usage

The propellant gaging function (PUGS) is required to monitor fuel and oxidizer consumption and/or loss. In its present form or with minor modifications, it will meet the functional requirements of the 700-day baseline planetary mission. The potential variation in the dimensions of the selected fuel and oxidizer storage tanks can influence the applicability of the sensor assembly in particular. For this reason (and others) there is some useful arguments for maintaining the present dimensions and using redundant tanks.

Four years of invaluable information and experience have been accrued to date in designing, developing, testing and fabricating PUGS, which can be applied directly to the program of upgrading areas of PUGS to meet the baseline mission criteria.

The Apollo PUGS has been used as a baseline for study to determine what areas require change in order to provide a PUGS that will satisfy the mission reliability, performance, safety and maintainability criteria for any extended mission program.

The Apollo PUGS is comprised of two fuel probes, two oxidizer probes (these may be increased by a factor of two for the baseline mission), and oxidizer control valve, a control unit and a display gage. All of the components will be located in the propulsion module, which is part of the EEM, with the exception of the display unit, which is in the Earth Entry Module and the Mission Module. The diagram in Figure 4.46 illustrates the probable PUG component location, interconnection and interfaces.

The PUGS function is critical to crew safety because of the need for very accurate control of engine firing duration which, in turn, establishes the resulting velocity vector changes. Its functions are therefore twofold, monitoring fuel and oxidizer remaining and responding to the "burn time" control signals from the guidance functions.

2. Reliability Predictions

Reliability predictions have been determined for the baseline PUGS for the Mars mission. A breakdown was made for all the major components of the system down to the module or circuit function level. In this way it became evident which areas of the system were most prone to failure. Additionally, it indicated whether or not a module or circuit function required improvement to meet the reliability requirements for the system. Tables 4.25 through 4.27 list the module or circuit function along with its failure rate for various major system components. The failure rates for each function remain the same regardless of mission duration. The Apollo mission on-time is 660 seconds, while the baseline mission on-time is expected to be much lower because of the recommended change in operational mode - about 150.5 seconds. Since this is so, the baseline mission duty cycle may be less than four times shorter than for the Apollo mission. Circuits not energized, will have negligible effect upon the components of the system, if they are not in the environment of the propellants. Reliability data supports this position and, as a result, reliability criteria have been met with existing circuit configuration. Further, it will be more than satisfactory for any of the planetary missions. The limiting factor is the sensor submerged in the propellant. The data indicates that the reliability of this function will probably exceed 0.991 per function; or 0.99992 probability that one of the two in each tank will function normally throughout the mission.

The only potential problem area will be the use of servo motors and motor generators, motor tachometers, potentiometers and gear trains, because of their use of lubrication in their bearings. In addition to the mission time of 700 days and a storage time prior to launch of possibly two to three years, the bearing lubrication may migrate out of the bearings to some extent. Selection of the proper bearing lubricant for the extended mission would be in order, and does not pose a significant problem.

A detailed failure mode and effect analysis was performed and may be found in Reference 4.5-5.

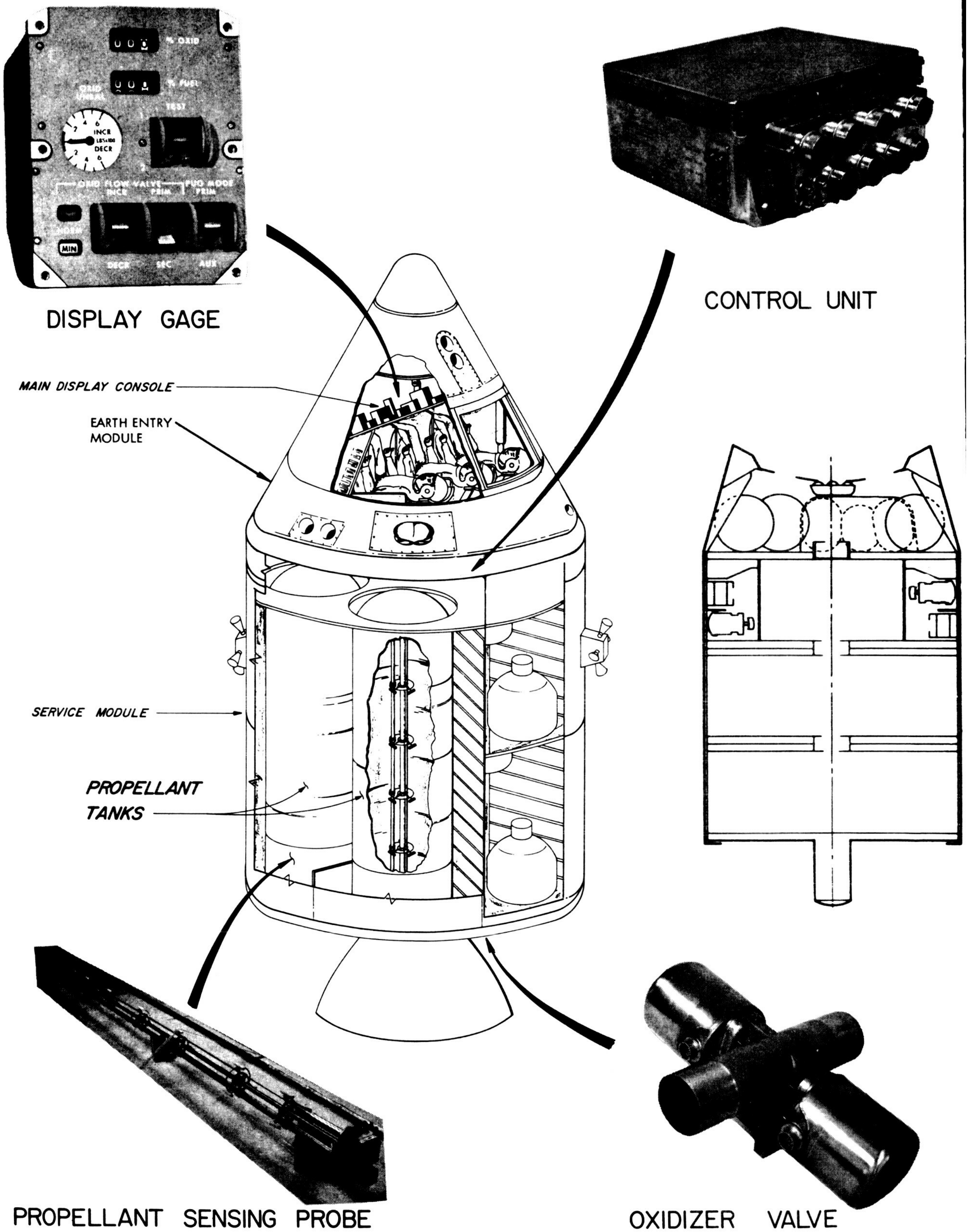


Figure 4.46. Component Interconnection and Interface

Table 4.25 Failure Rate Estimates, PUGS Control Unit

Module	Description	Percent/1000 hrs.
A100	Flag servo ampl.	1.5212
A101	Display oxid. ampl.	0.5280
A102	Display fuel ampl.	0.5280
A103	Unbal. servo ampl.	0.6594
A104	Pri. fuel ampl. tank 1	0.2424
A105	Pri. oxid. ampl. tank 1	0.4747
A106	Pri. oxid. ampl. tank 2	0.4687
A107	Pri. fuel quad. ckt. tanks 1 and 2	0.0409
A108	Pri. fuel ampl. tank 2	0.3124
A109	Discrepancy ckt.	1.3399
A110	Pri. power supply	0.8876
A111	Aux. power supply	1.3051
A112	6 kc osc.	0.0369
A113	D/A or gate fuel	0.3899
A114	D/A or gate oxid.	0.4695
389850	Gear train assy., including motors and pots.	10.6560
A116	Aux. oxid. ampl.	0.3527
A117	Aux. fuel ampl.	0.3340
A118	Empty and full adj. ckt., tank 1 and 2	0.1724
A115	Motor control caps, etc.	0.2334
	Grand Total	20.9531

Table 4.26 Failure Rate Estimates, PUGS Fuel and Oxidizer Sensors

System	Description	Percent/1000 hrs.
Primary	Fuel Sensor 1, storage tank	0.0126
Primary	Fuel sensor 2, sump tank	0.0126
	Subtotal	0.0252
Primary	Oxidizer sensor 1, storage tank	0.0146
Primary	Oxidizer sensor 2, sump tank	0.0146
	Subtotal	0.0292
	Primary system Total	0.0544
Auxiliary	Fuel sensor 1, storage tank	0.0166
Auxiliary	Fuel sensor 1, sump tank	0.0166
	Subtotal	0.0332
Auxiliary	Oxidizer sensor 1, storage tank	0.0198
Auxiliary	Oxidizer sensor 2, sump tank	0.0198
	Subtotal	0.0396
	Auxiliary system Total	0.0728

Table 4.27 Failure Rate Estimates, PUGS Display Gage

Section	Description	Percent/1000 hrs.
1	Oxidizer display servo	1.732
2	Fuel display servo	1.734
3	Unbalance display servo	1.736
4	Flag display servo	1.740
5	Test circuitry	1.721
6	Valve switch	0.120
7	Oxidizer flow switch	0.062
8	Control switch	0.180
	Total	9.027

Table 4.28 Failure Rate Estimates, PUGS Oxidizer Valve

Section	Description	Percent/1000 hrs.
1	Primary valve	1.818
2	Secondary valve	1.818

3. Availability Analysis

Although the PUGS function meets the required goals through operations control, it would be well to note that application of the availability concept could improve the mission safety through repair and maintenance of potentially weak links. The data given in Figure 4.47 indicates that the weakest link in the function is the Control Unit and further, Table 4.25 indicates that its weakest link is the gear train assembly. From Figure 4.48, the gear train is identified as a separable assembly which can be replaced as a unit, if the control unit case is modified to permit ready access. However, the duty cycle is too low to expect the need for replacements during the engine firing even if all the lubricant was lost due to migration.

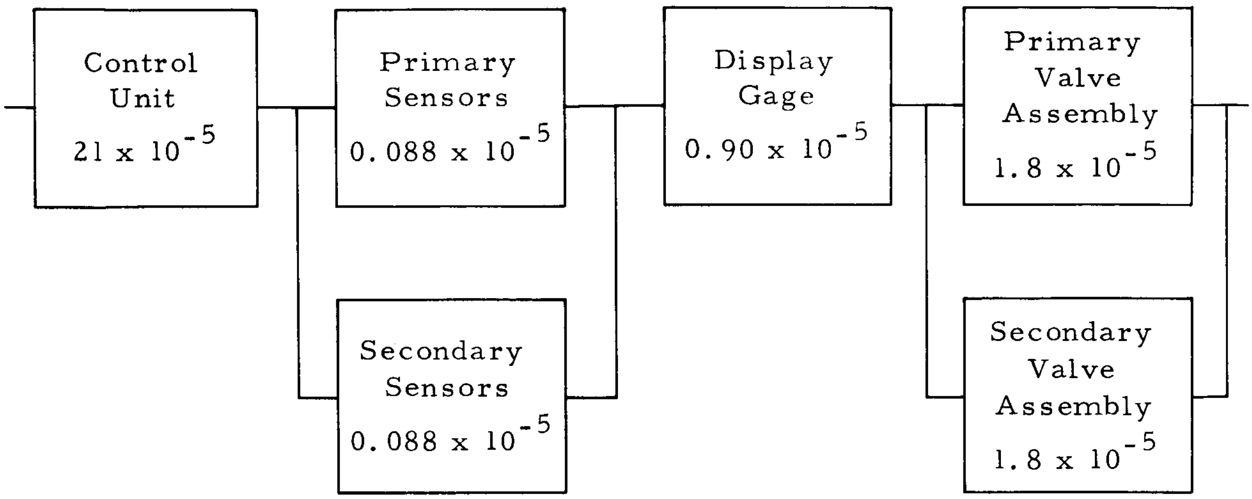


Figure 4.47. Reliability Logic, Propellant Gaging Function

An adequate lubricant can be selected for the extended mission application to preclude the possibility of operation with dry bearings. Prior to a mission, the checkout testing for the components (if not the system), will involve test durations far in excess of mission on-time requirements.

One other failure mode that deserves consideration is that of a PUGS fuse being opened as a result of external electrical transients. The PUGS fuses are now located in the hermetically sealed control unit, which, in its present configuration and location, is not maintainable. The minimal change to correct this situation would be to locate the control unit, preferably to the EEM, and provide a gasketed access cover for replacement of blown fuses. Other elements in Table 4.30 contribute to its high failure hazard, 3 or 4 modules are weaker than the others and could be replaced if provisions for gear train replacement was provided.

4. Conclusion and Recommendations

The data indicates that the PUGS will produce satisfactory performance on the baseline mission. The weakest link, the sensors, will exceed 0.99992 probability of functioning correctly. Since the pertinent Apollo and planetary mission environments are essentially the same, with the exception of mission duration, the principal environment of concern is that of propellant exposure for the sensors and valve.

A review was made of the Apollo Qualification Test Programs specifically for propellant exposure information. Each of four oxidizer valves had been exposed to N_2O_4 for $\simeq$ 130 hours for a total of 5200 valve unit hours, and no signs of deterioration were noted as a result of exposure. Each of four oxidizer and four fuel sensors were exposed for ≃ 2800 in their respective propellants, for a total of 11, 200 fuel or oxidizer sensor unit hours.

The greatest potential problem area is confined to solder joints and associated feed-throughs in the sensing probes. Although a considerable amount of effort and testing has been expended in this area, further investigation and tests are warranted in these areas to provide adequate assurance for the greatly extended mission duration since they will not be maintainable during a mission.

4.5.5 Propellant Control Function

The propellant control function provides the means of moving both fuels and oxidizer from the storage tanks (passive or positive expulsion) to the engine proper. This function is required to service:

(a) Reaction control engines

(b) Gravity control engines

(c) Main propulsion engines

Once the propellants leave the tanks, the method of control is very similar for all three functions, although the method of expulsion differs.

A study of the Apollo configurations, as applied to the baseline mission, indicates that the same configuration will be directly applicable to any of the potential planetary missions and spacecraft configurations, with minor modifications to fit the specific application and selected operational concept.

The RCS and gravity control systems are expected to be enough alike to discuss and analyse them together; however, the main propellant control is handled separately.

1. The RCS and Gravity Control Propellant Feed Function

The reliability logic diagram for this function was given in Volume II of this report. The reliability assessment is presented in Table 4.29 The assessment indicates that the system will meet the mission functional requirements, but in the Apollo configuration, it displays two weak links that should be investigated. The failure hazard should be reduced to achieve a P_s of over 0.99993. The weak links are the helium supply check valve and the helium storage tank - in that order.

Table 4.29 Reliability Assessment RCS and Gravity Control, Propellant Control Function

Component	Reliability Without Change	Contribution to Crew Safe Return	Suggested Corrective Action
1. Helium tank	0.99930*	0.999994	Gate valve and redundant supply for 1 of 4
2. Helium fill valve and disconnect	> 0.999999	>0.999999	None required
3. Helium solenoid and regulator	0.999963	0.999963	None required
4. Helium supply check valves	0.99899*	>0.999999	Pressure relief during inactive periods
5. Burst disk	> 0.999999	0.999999	None required
6. Helium plumbing (leaks)	0.99998	>0.999999	Pressure relief
7. Fuel fill valve cap	> 0.999999	>0.999999	Not required
8. Fuel plumbing (leaks)	0.99998	>0.999999	Pressure relief
9. Fuel valve and bellows	0.99997	0.99997	Not required
10. Fuel filter	> 0.999999	>0.999999	Not required
Total for each function	0.9983	0.99993	

*Weak links

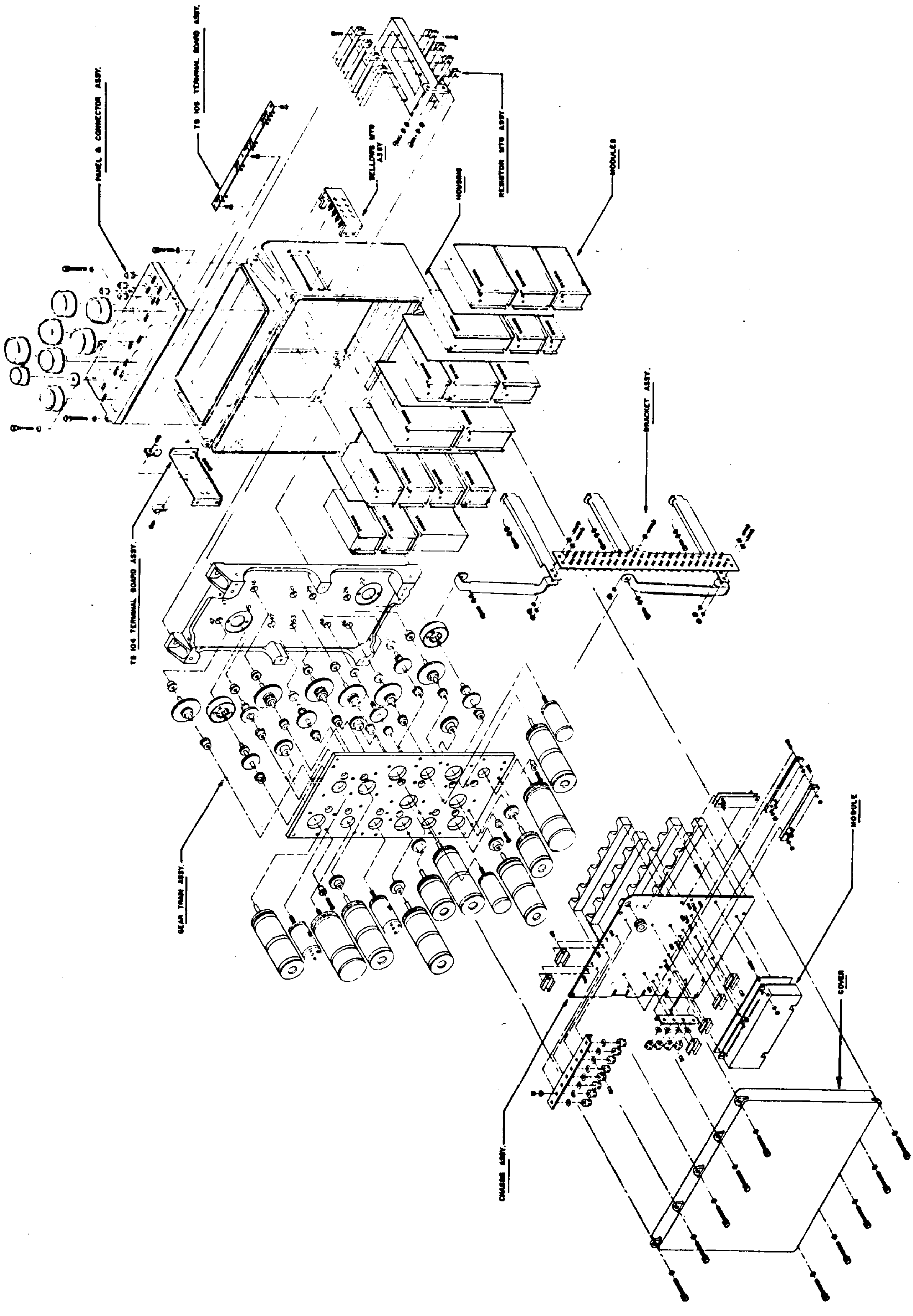


Figure 4.48. Exploded View, Control Unit

The helium supply check valve problem was relieved by a change in operational concept. By relieving the pressure in the system during the long artificial gravity coast phases, the operational duty cycle was reduced by a factor of about 19. This decreased the probability of a failure to less than 1×10^{-6}. This change in operational concept is not expected to introduce any problems; but rather, it has further decreased the probability of failure in other components of the function.

The helium tank could fail in two modes by leakage and by burst. The effects of leakage is reduced to insignificance through use of a gate valve at the outlet, and by including a redundant source of helium. The probability of bursting a tank is virtually eliminated by increasing the tank wall thickness by about 10 percent.

2. The Main Propellant Feed Control Function

The reliability logic diagram for this function is presented in Figure 4.50 and the resultant assessment is presented in Table 4.30 The weak links in this function are similar to those in the RCS feed system. The weak link seems to be the tankage, but available engineering data indicates that this can be corrected by increasing the tank wall thickness. The resultant weight change would be insignificant.

As indicated in the referenced table, addition of a redundant tank has about the same effect on weight and an appreciable increase in the function effect on P_s; whereas the function contribution to P_s could be as low as 0.989, the redundant tank and feed function increases this to more than 0.993. This is a practical alternative since the required fuel can be divided between the two redundant functions.

A further increase in P_s is achievable by isolating the H_e pressurant at the tank and venting the feed lines. This eliminates the major failure mode and will probably reduce the failure hazard to less than 1×10^{-6} per mission.

Table 4.30 Main Propellant Feed Function, Contribution to Crew Safe Return Analysis

Sub Function	Logic Block	Duty Cycle	2×10^{-6} (Apollo & AAP)	Contribution to P_s*
Helium Supply	R_1	16,602.8 hrs	0.0220	0.999635
	R_2	4.0 hrs	0.0020	
		4.0 hrs	2.9762	0.9999
	R_3	16,602.8 hrs	0.0020	0.9999
	R_4	7 cyc	0.0586	0.9999
	R_5	7 cyc	16.7×10^{18}	0.9999
	R_6	16,602.8 hrs	0.000036	0.9999
	R_7	16,602.8 hrs	2.0×10^{6}	0.9999
	R_8	16,602.8 hrs	4.0×10^{6}	0.9999
	R_9	16,602.8 hrs	0.04761	0.9992
Subtotal				.9988
Propellant Supply	R_{10}			
Tank prop.		16,602.8 hrs	0.0850	0.994356
Valve, vent.		4.0 hrs	0.0020	0.999999
Valve, fill/drain		16,602.8 hrs	0.0020	0.999934
Heat exchanger		16,602.8 hrs	0.1000	0.99668
Bellows		16,602.8 hrs	0.0119	0.999605
Subtotal				0.991
Function Total				0.9894
Add to propellant Supply and purge capability				
Tank, prop.		16,602.8 hrs	0.0850	0.994356
Valve, vent		4.0 hrs	0.0020	0.999999
Valve prop. isol.		16,602.8 hrs	0.0200	0.999987
		7.0 cyc	0.01 cyc	0.999999
Subtotal				0.9943
Revised Function Total				0.993

*These values are known to be very conservative.

4.6 THE MAIN PROPULSION ENGINE

Note: Much of the data used in this section was provided by the Aerojet General Corporation through Reference 4.12 and 4.13.

4.6.1 Functional Description

The main propulsion engine was assumed to be the Apollo Service Module Engine where one, two, or three were used for the baseline mission. For usage details see Volumes I and II of this report. The baseline mission study indicated that this was one of two possible engines that could be used, the LEM descent engine (LMD) or the SM engine. The SM engine was selected for this study because of the preponderance of accurate data available from the Apollo program at SD the data is indicative of either engine capability.

The main requirements of the engine are to provide the necessary thrust for midcourse correction during the transplanet phase, trajectory correction during planet approach, midcourse correction during the trans-earth phase, and retrofiring for earth approach and reentry. The total engine firing required for the complete mission of about 700 days is an average of seven cycles for a maximum total duration of 155 seconds.

The subsystem schematic is given in Figure 4.49 along with a list of functioning components.

4.6.2. Reliability Analysis

Reliability logic diagrams are presented in Figure 4.51 for engine start and steady state operations and Figure 4.52 for engine shutdown operations and coast periods. Associated with each component or assembly identified in Figures 4.51 and 4.52 are all of the various parts associated with its construction. The bipropellant valve has been separated into four valve assemblies, each consisting of an actuator and the associated pair of ball valves (one for fuel and the other for oxidizer), shafts, seals, springs, housing, and other related parts. The gimbal actuators for pitch and yaw are considered separately. In each actuator, the redundant motor-clutch assemblies are individually identified. There are two redundant electrical harnesses assumed.

The reliability logic diagrams shown in Figures 4.51 and 4.52 provide a reasonably rigorous basis for calculating the engine reliabilities. A more detailed set of logic diagrams, for the lunar mission, are presented in Reference 4.6-2.

The engine contribution to P_s should be assessed in both the active and passive phases since the engine will be dormant for a major part of any planetary mission.

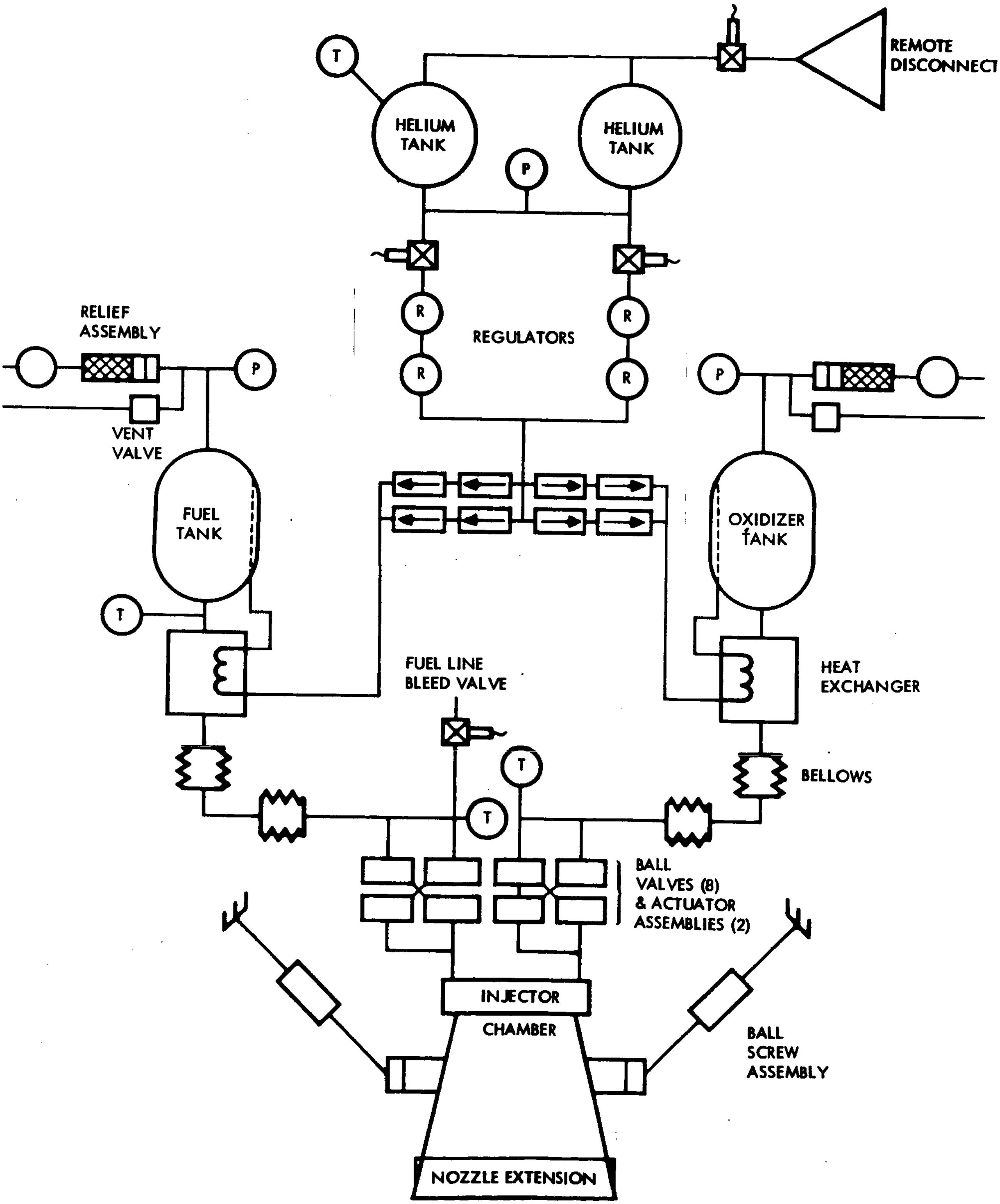


Figure 4.49. Subsystem Schematic

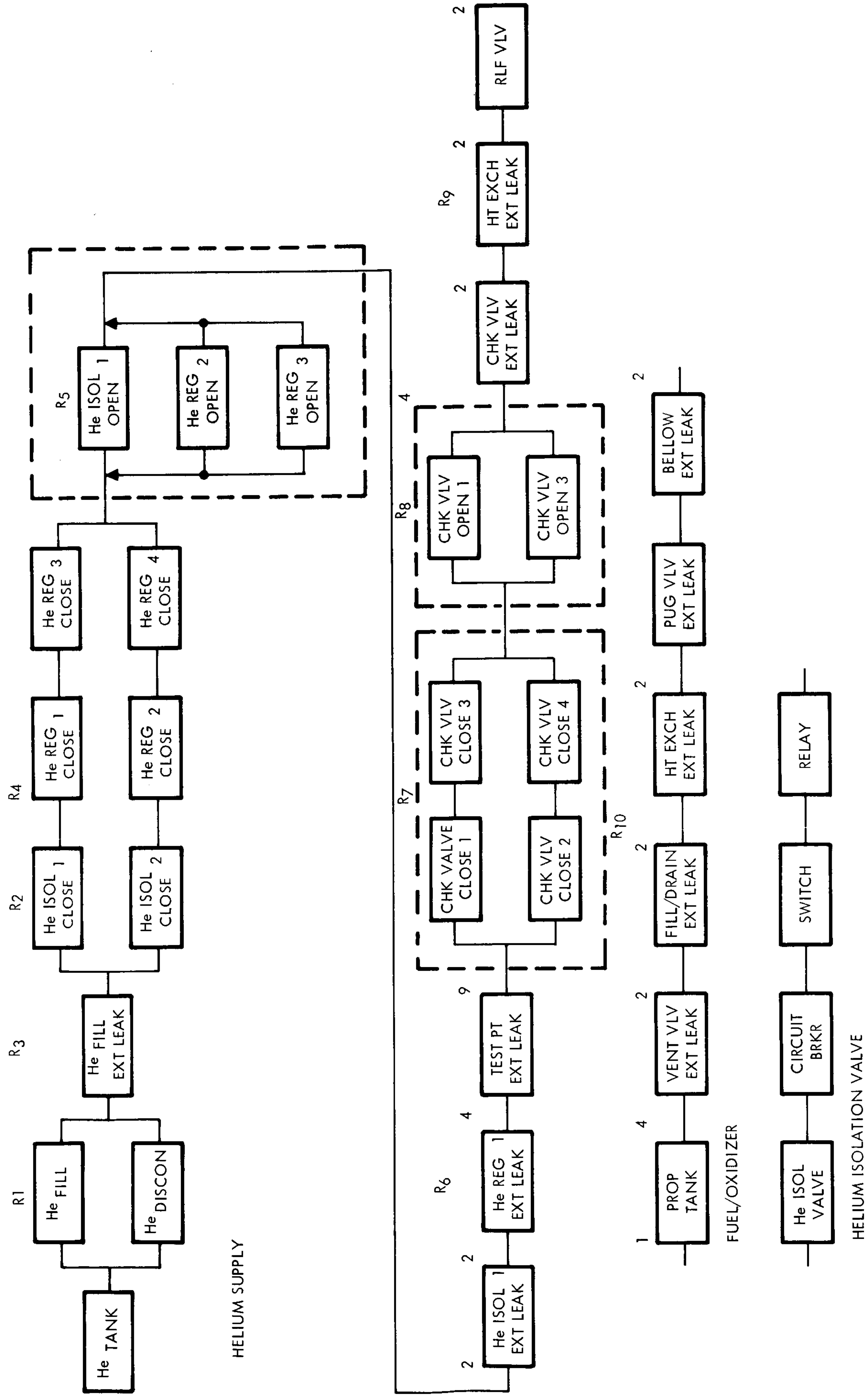


Figure 4.50. Main Propulsion Subsystem, Reliability Logic

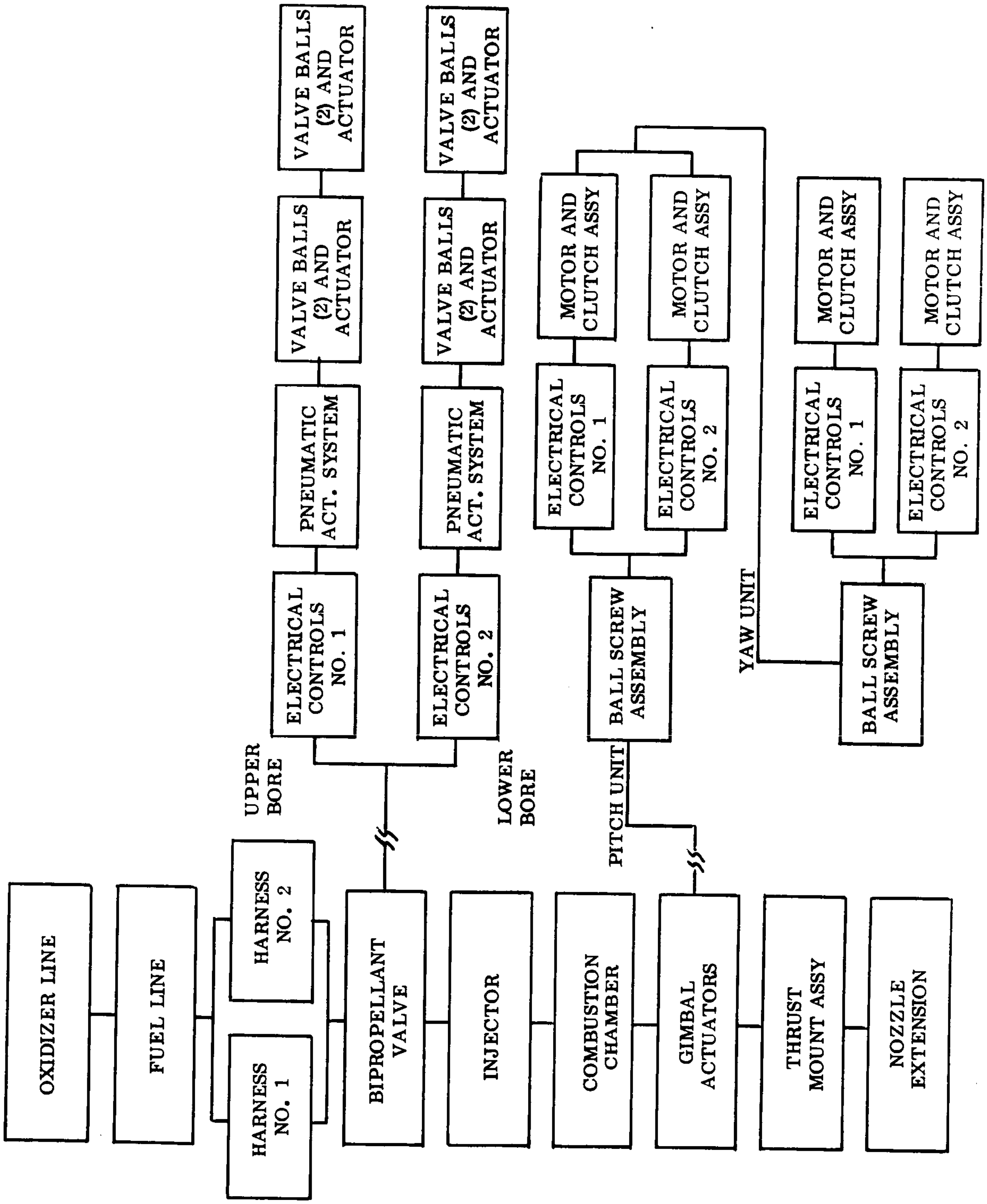


Figure 4.51. Reliability Logic Diagram for Engine Start and Steady-State Operations

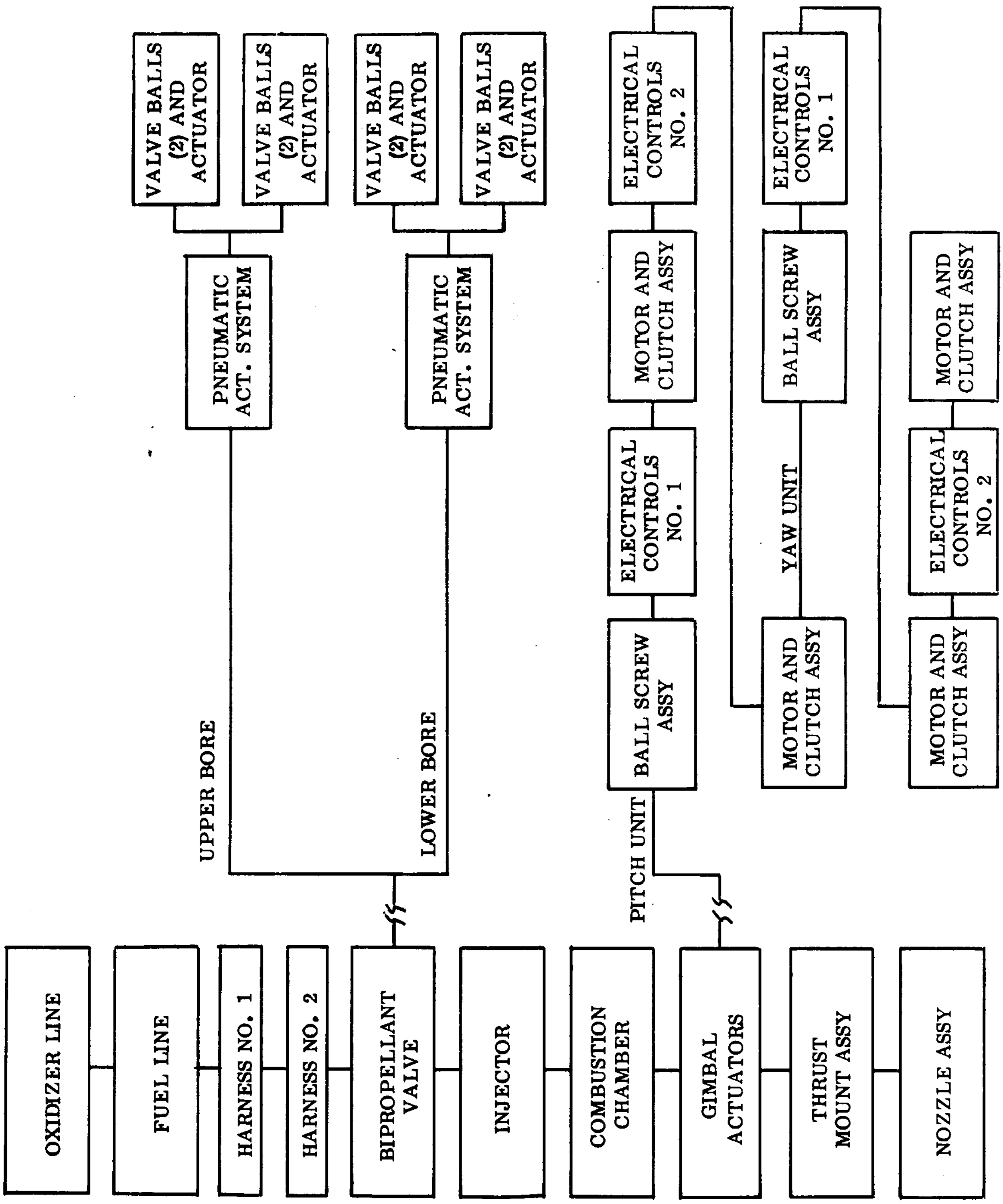


Figure 4.52. Reliability Logic Diagram for Engine Shutdown Operations and Coast Periods

The Active phases include the time from fuel system pressurization through the burn to the time pressure is relieved and the feed system is purged. This phase is normally considered the most hazardous.

Reliability estimates given in Reference 4.13 for the lunar and baseline missions are presented in Table 4.31 These reliability estimates provide a basis for estimating reliabilities for the Mars flyby mission, which is related to the ratio of burn time, 155 for the baseline mission vs. 600 for the Lunar landing mission. (The longer burn time for lunar mission is created by the large ΔV required for escape from Lunar orbit).

The hot-fire engine reliability for the baseline mission would be on the order of 0.996 without changes. This level of reliability is probably adequate. However, if it is deemed necessary to increase the engine hot-fire reliability, the combustion chamber reliability may have to be improved before a significant increase in engine reliability could be achieved. A review of the hot-fire test history of engine combustion chambers indicates only two failures. In both cases, the chambers were tested to failure - one for a duration of 1956.4 seconds, the other for 2392.4 seconds. Both failures were of the wearout type, and a value of 0.99999992 can be calculated as the estimated probability of no wearout for the baseline mission. There were a total of 25 chambers tested for durations of at least 750 seconds without the occurrence of any random-type failures. These data can be used to obtain a median value for the MTTF of about 27,000 seconds which result in a demonstrated random reliability of 0.9943 associated with the 115-second hot-fire duration specified for the planetary flyby mission. This test experience indicates an adequate (perhaps overly conservative by reference 4.13) design with respect to wearout. The demonstrated random reliability is quite low due to lack of data. The true random reliability is known to be significantly higher, but this cannot be demonstrated without accumulating more test data. Since no random failures have occurred up to the present time, there is no reason for recommending any change in design. The current combustion chamber (and all other engine components as well) can be accepted without change for the hot-fire portion of the Mars flyby mission.

The Passive Engine phase of the proposed Mars flyby mission involves consideration of the effects of the prolonged coast period (about 700 days) on coast reliability. Reliability estimates are given in Reference 4.13 along with a qualitative appraisal of the anticipated effects of baseline mission. As an aid in making the required qualitative judgments, it was found helpful to perform a failure mode-cause-effects analysis (FMEA) of each component. The FMEA analysis is presented in Reference 4.13.

There is a definite controversy over the effects of long-term exposure to the space environment, and there is little data to verify any position. So

Table 4.31 Estimates of Component Reliability for all Periods of SPS Engine Firing

Component	Engine Hot-Fire Reliability	
	Lunar Mission	Baseline Mission
Thrust mount	0.999052	0.999755
Nozzle extension	0.998887	0.999712
Combustion chamber	0.993596	0.998342
Injector	0.998959	0.999731
Propellant lines	0.996634	0.999623
Gimbal actuator (each)	0.998541	0.999623
Pneumatic actuation system*	0.989623	0.997309
Bipropellant valve	0.999524	0.999877
Single Engine Total		0.996

*The reliability figures given for the Pneumatic Actuation System are applicable to each of the two pneumatic systems which are functionally redundant; that is, failure to supply pneumatic actuation pressure.

far, there have been no failures attributable to this effect. The FMEA indicates the potential effects and, in each case, protective measures can be taken to prevent the problem. Since this is true, this study will be based on that assumption; however, this represents one of the major problem areas requiring a full scale test program.

4.6.3 Availability Requirements Analysis

The reliability analysis shows that the Apollo engine will meet the baseline mission hot-fire requirements with a higher P_s than for the lunar mission, and with a large margin for error. However, the long vacuum-cold soak and the associated unknowns make it desirable to consider ways of augmenting this function and reducing the potential effects of the unknown.

Four solutions seem apparent and are not mutually exclusive:

a. Use of multiple engines

b. Limited maintenance and repair

c. Use of heaters on the fuel lines and the injector

d. Relieving pressure and purging fuel/oxidizer lines after use

Maintenance is discussed in paragraph 4.6.4, the last two require no explanation other than to indicate that the proper application of both would negate several of the more prominent failure modes, both in the engine proper and the feed function.

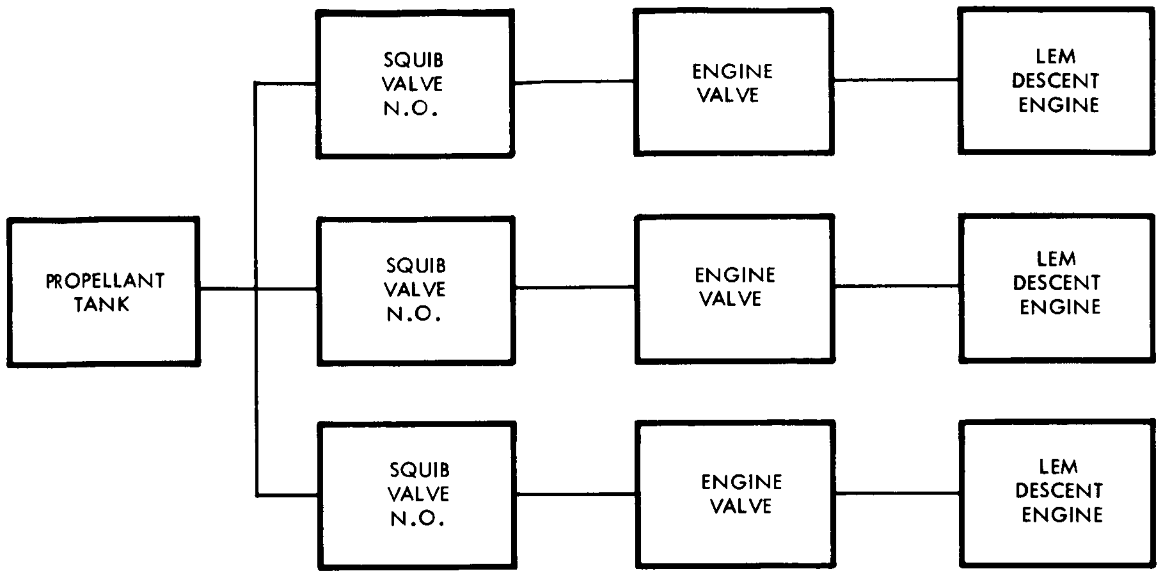
The use of a redundant engine configuration provides a significant increase in reliability relative to crew safety and mission success by elininating a critical single-point failure condition and incorporating additional flexibility in the number of acceptable operational modes. However, this does increase system complexity and adds weight.

This evaluation is concerned with the ramification of utilizing three Apollo engines (LMD) to provide a one-engine-out capability, thereby effecting greater propulsion flexibility and reliability during missions.

The configuration involves a common manifold with individual propellant shutoff valves for each engine to provide isolation in event of engine failure (Figure 4.53(a)). The isolation valves are assumed to be normally open and explosively operated closed and also serve as a backup to an engine valve failure open condition, Figure 4.53(c). The failure hazard of inadvertent operation, closed by the explosively operated valve, would effect loss of one engine leg; however, the subject configuration is capable of two out of three engine leg operation as in Figure 4.53(b).

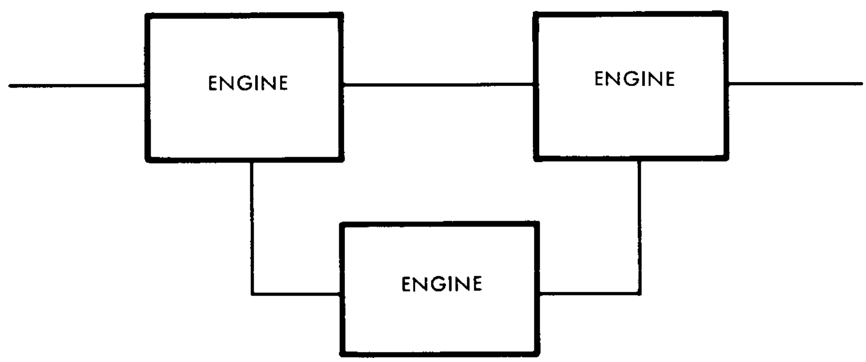
Review of possible gimbal requirements considered two potential configurations; (1) A single gimbal platform incorporating the three engines, and (2) individual gimbal assemblies for each engine.

The Individual Gimbal Assembly Concept results in a series relationship of the gimbal with its respective engine and a two out of three total capability Figure 4.54(b). The loss, therefore, of one gimbal assembly causes only the loss of use of one engine. This configuration concept requires three platforms with increased installation complexity and also presents formidable interface problems with the SCS and EPS systems. For accurate SC control, the three gimbal assemblies must achieve positive phase angle

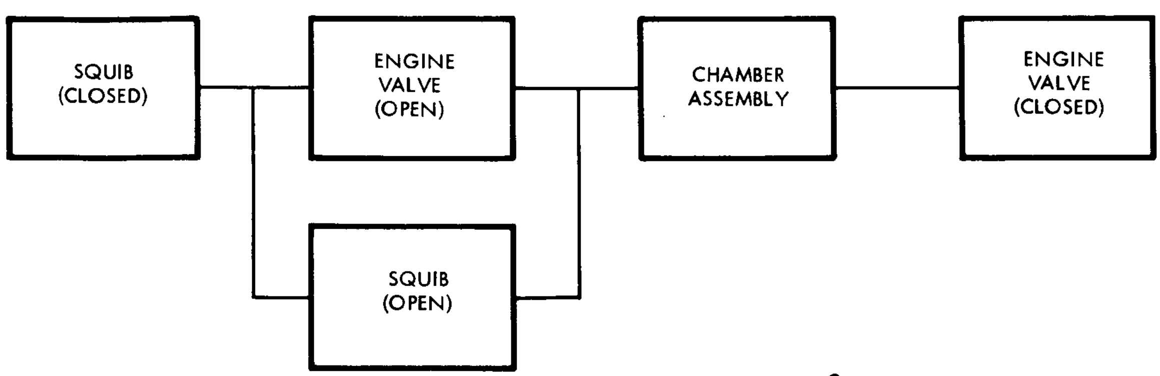


FUNCTIONAL DIAGRAM A

REQUIREMENT: ANY TWO OF THREE ENGINES



RELIABILITY LOGIC DIAGRAM B



EACH ENGINE RELIABILITY LOGIC DIAGRAM C

Figure 4.53. Engine Configurations Logic Diagram

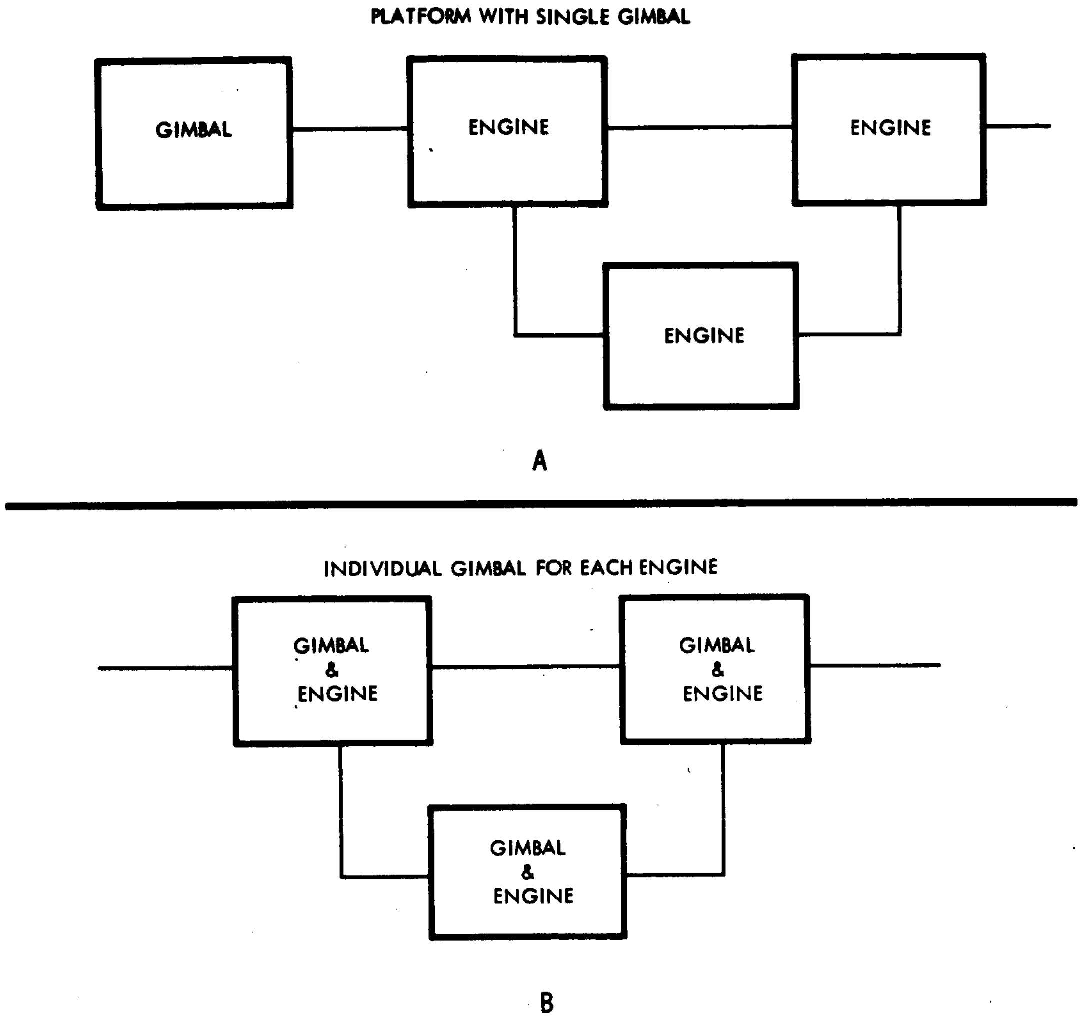


Figure 4.54. Gimbal, Logic Diagrams

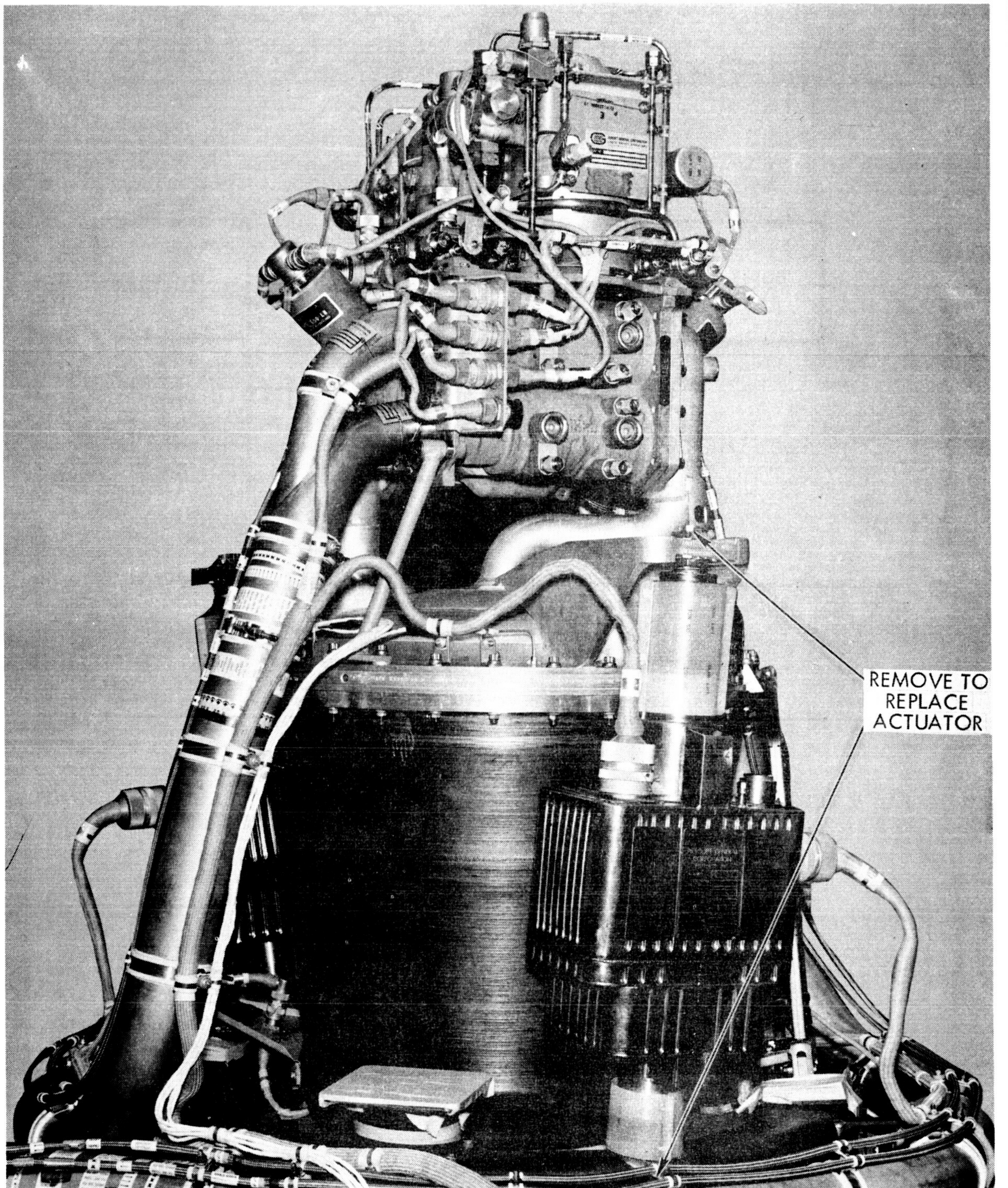


Figure 4.54A. Photo, Main Engine Showing Replaceability of the Gimbal Actuator

interlocking including accurate intelligence relative to gimbal responses. The EPS system would also be affected by the increase in power loading requirements.

The resultant P_s for the proposed configuration would increase from single engine 0.996 to over 0.99999, discounting the effects of storage. The effects of storage are overcome by sealing off one engine and using it in the event of failure of the others.

The Single Platform Concept provides for minimal installation and functional problems when considering the A and SCS and EPS interfaces. This platform would be similar in configuration as currently incorporated on Apollo excepting mounting provisions for the three engines. It also presents a potential single point failure mode that could cause loss of use of the engines (Figure 4.54(a)); therefore this would necessitate extremely high reliability. However, it is probably the most reliable of the two mounting concepts, all factors considered.

4.6.4 Maintainability Analysis

Although maintenance and repair operations are not necessarily recommended for the main engine, there is good reason to believe some of the more prominent failure modes could be remedied by some comparatively simple replacement actions. From Table 4.31 the weakest link is found to be the gimbal actuator (pneumatic). From Figure 4.53 it is evident that the unit could be replaced in toto by removing and replacing:

2 electrical connectors

2 lock nuts

2 retaining nuts

The tasks have been analyzed and are known to fit within the astronaut constraints envelope, even under EVA, provided he has a simple restraint; this without change to the present design. The photo also reveals that, although it is a somewhat more complex operation, the complete valve assembly could be replaced if the two welded joints were replaced with a flange connection.

4.6.5 Conclusions and Recommendations

The study indicates the Apollo type engines will more than meet the hot-firing requirements for the planetary flybys, basically, because the burn time is about one-fifth of the present design level. The effects of the long coast periods and deep space soak have not been evaluated sufficiently to draw any safe conclusions. The available data seems to indicate that few problems should be expected since no failures were encountered in space attributed to this cause. Maintenance of sensitive components is possible

Table 4.32 Estimates of Component Reliability for Coast Periods

Component	Lunar Mission Coast Reliability	Effects of Extended Coast of Mars Mission on Coast Reliability	
		None	Potential Decrease
Thrust mount	1.000000	X	
Nozzle extension	0.994028	X	
Combustion chamber	0.993348		X
Injector	0.998751		X
Propellant lines	0.995960		X
Gimbal actuator (each)	1.000000		X
Pneumatic actuation system	0.995022*	X	
Bipropellant valve	0.999917	X	

*This value is applicable to each of the two redundant pneumatic actuation systems

and minimum redesign may be desirable, however, M and R is not required to meet the objective. Use of multiple engines can reduce the chance of failure to less than 1×10^{-5}.

Areas recommended for further study include:

1. A study of metal creep seems warranted because of the long mission duration. Of particular concern are the tanks and lines in the pneumatic actuation system that are continuously exposed to the high pressure GN_2; the design should be conservative enough to assure that the creep strength of the materials are not exceeded. Another area of concern is the possible stretching of bolts used in intercomponent joints; the bolt materials should be analyzed to determine whether a design change is required; retorquing is required during coast; or no change is required.

2. Determine protection requirements against meteoroids - both for the engine proper and the nozzle extension

3. Further studies are required of the effects of prolonged exposure of the non-metallics to the propellants and the hard vacuum of interplanetary space. Data regarding long-term exposure in space should be obtained from existing satellites, if possible.

4. An evaluation of the effects of hard vacuum on the binder used in the combustion chamber. If binder sublimation is found to be a problem, then a method for retaining low-pressure GN_2 within the chamber during coast can be developed and evaluated.

5. Redesign of some component interfaces may be necessary to facilitate parts replacement. With the present system design it would be impractical to change some components in space.

6. An improvement in the bipropellant and valve propellant sealing may be required to avoid propellant loss, evaporative freezing, and possible explosion during ignition subsequent to the long coast period; or, the inclusion of a gate valve and purge may be a better alternative.

4.8 GUIDANCE AND NAVIGATION

Note: Much of the data used herein has been provided by the A. C. Electronics Division of General Motors Corp. through Reference 4.14.

4.8.1 System Functions

The Guidance and Navigation (G and N) system is employed to provide a means of determining the spacecraft position and velocity vector with respect to the planets of interest. Two basic operational modes of navigation are possible; local primary and earth primary via the communications link (Section 2.1). Former studies indicate that earth primary is the optimum mode for all mission phases except in the immediate vicinity of planets or during earth reentry.

The G and N system is expected to be composed of an optical navigation subsystem, a guidance computer, and an inertial measurement unit. These same functions are required for the Apollo mission. Therefore the associated components formed the basis of this analysis. The optical subsystem possesses the ability of measuring the angle between a star and planet center, or some landmark thereon, and the capability of measuring the apparent angle subtended by a planet (stadiametric). The guidance computer (AGC) is to have the capability of performing the necessary calculations to relate the optically measured data to the position and velocity of the spacecraft; and when required, to calculate the vehicle course corrections required. The inertial measurement unit (IMU) provides a reference coordinate system and monitors the applied accelerations to the vehicle.

The functional logic for the Mission Module G and M functions are given in Figure 4.55. The functional logic for the Earth Entry Module system is given in Figure 4.56. It will be noted from these examples that the different modules impose somewhat different requirements. The earth entry modu module functional requirements are similar to Apollo in all respects and the Apollo system is expected to meet those needs or exceed them without change. The mission module system is to facilitate velocity corrections only and function under different circumstances. This is the system studied in depth herein, the recommended operational concept is justified in Section 2.1 of this report.

4.8.2 Reliability Analysis

The capability of the Apollo G and N system, using present operational concepts to meet the above requirement, can be grossly evaluated by considering the entire G and N system ON (energized) for the maximum duty cycle time of 825.16 hours, as shown in Volume I of this report. The total G and N system failure rates are 895.6 and 77 failures per mission hours

Table 4.43. Estimated Reliability for the Electric Power Source, Thermalelectric Backup System (16,000-Hour Duty Cycle Assumed)

System of Component	SiGe System	
	$\lambda \times 10^{-7}$	Reliability
Heat Source		
Isotope capsules	0.53	0.99907
Heat exchanger	1.2	0.9979
Shield	0.04	0.99993
Primary Coolant System		
Piping	0.4	0.9993
ECU		0.99996
Converter, electrical	1.0	0.9982
Heat Rejection System		
Piping	0.4	0.9993
ECU		0.99996
Radiator		
Performance	2.13	0.9963
Meteoroid Requirements		0.9957
Capsule safety ejection	0.95	0.9984
Across-the-line pump	2.4	0.9958
Performance reliability		0.984
Total Reliability	9.05	0.980

Table 4.44. Redundancy Recommendations, 5 kWe TE Power Source

Recommended Design Action	Two-Year Mission	Emergency Only
1. Redundant separate-source pumps in primary 2. Two active converter-heat rejection segments, plus one standby segment 3. Shutoff valves on heat rejection		
Performance reliability, TE system	0.9902	0.9999+
Required meteoroid non-puncture reliability per radiator segment	0.997	Depends on design
	0.99	0.999+

Power Return — The distribution system is normally a two-wire grounded system for DC and single-phase AC loads, and four-wire grounded system for three-phase AC loads; i.e., wire and buses should be employed as the return path for electrical currents rather than the spacecraft structure. The system negative and neutral should be grounded at one point only in the spacecraft and not be interrupted by any control or switching device.

The recommended DC and AC power characteristics for the baseline mission is as follows, as sensed at the load: (Reference 4.9.2-1).

a. DC Power

1. Steady-state voltage limits:	27.5 plus or minus 2.5 volts
2. Transient voltage limits:	21 volts to 32 volts with recovery to steady-state within one second
3. Ripple voltage:	1 volt peak-to-peak 30 to 15,000 cps

b. AC Power, if an AC power distribution system is used

1. Phases:	Three phases — displacement 120 degrees-tolerance, plus or minus two degrees
2. Nominal voltage limits:	
Steady-state	115 plus or minus 2 volts rms (average of three phases measured line to neutral)
Transient	115 plus 35, minus 65 volts rms recovering to 115, plus or minus 10 volts rms within 15 milliseconds and steady-state within 50 milliseconds.
Unbalance	2 volts rms (worst phase from average)
Modulation	0.5 percent maximum

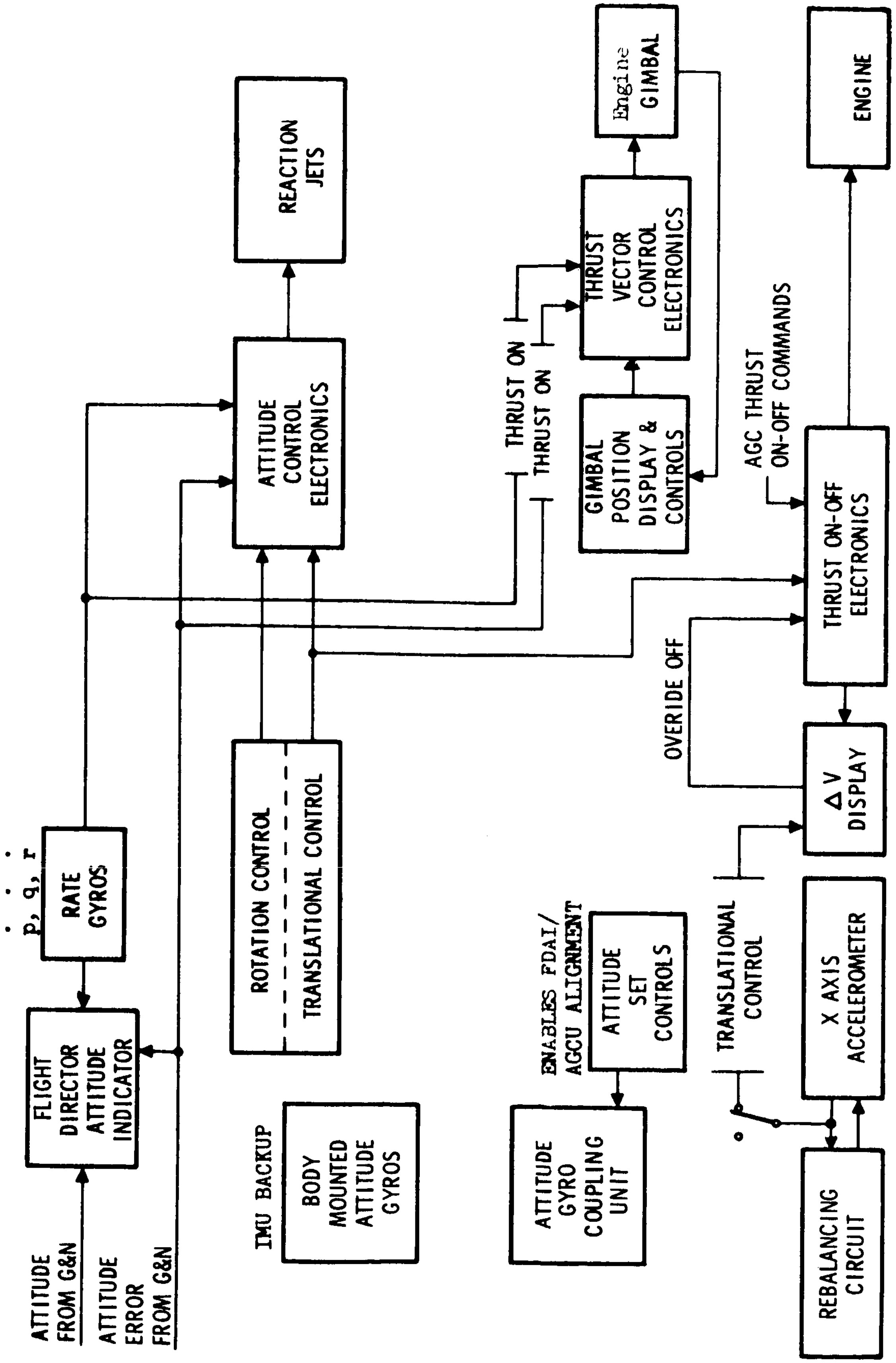


Figure 4.55. G&N ΔV Modes

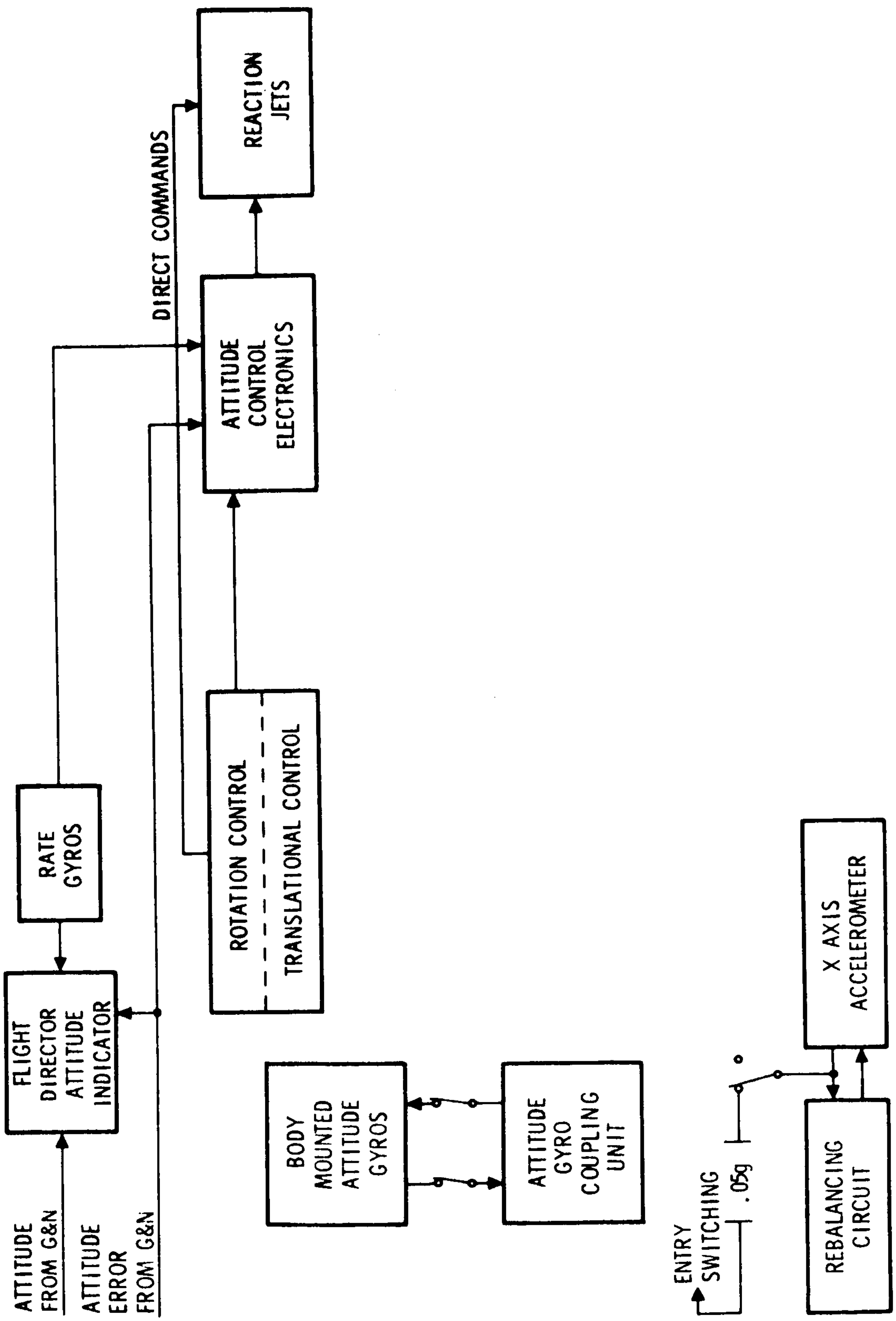


Figure 4.56. G&N Entry

(Reference 4.15) for ON and STANDBY usage, respectively. Applying the maximum duty cycle time to the ON failure rate, and the remainder of the mission time to the standby failure rate, the functional reliability becomes 0.478 and the standby reliability becomes 0.294 for an overall G and N reliability in the mission of 0.141. The reliability logic is given in Figure 4.57. Three spare G and N systems would be required to bring the ON reliability up to 0.9939, while four spares would be required to bring the "standby" reliability up to 0.9920. The overall G and N reliability would then exceed the required value of 0.99.

This approach is obviously pessimistic since all of the G and N equipment is not energized during every operating interval, and greater reliability gains can be realized by sparing at a lower equipment level than whole G and N systems. However, an upper bound has been established, and it does point out that an exorbitant amount of extra equipment would have to be carried by attempting to achieve safety through redundancy or spares at the component level.

The greatest reliability penalty results from the long standby periods when the G and N system is not required to perform any guidance, navigation, or control functions. During these intervals, the computer clock electronics and the inertial instrument suspension and heater circuitry are operative under present apollo operational concepts. It would appear quite difficult to improve the STANDBY failure rate enough by simple redesign, and the sparing of heater and suspension assemblies inside the IMU would seem difficult and very time consuming, at best. A more promising approach would be to change the operational concept so that all G and N equipment can be de-energized during STANDBY periods.

This approach greatly relieves the reliability problems. However, the problems of updating computer time to the required accuracy and possible inertial instrument performance degradation due to the lack of temperature control and suspension must be fully investigated. Presently the computer time is updated during the boost because of overloading of required functions at liftoff, so the operational concept is feasible. The effect on accuracy that the time update has to be performed to successfully accomplish mission objectives and what accuracy can be provided over great transmission distances should be investigated. Considerable study has already been performed on the effects of cool-down and loss of suspension on the inertial instruments, the results are encouraging and indicate the feasability of meeting the requirements of the baseline mission.

The effort herein was concentrated on isolating the weak reliability links in the operational equipment and determining the possible courses of action to bring the functional reliability from 0.478 to over the 0.99.

Since the reliability goal is fixed, any error or uncertainty in equipment usage will be directly reflected in the amount of redesign, spares, or redundancy required. Therefore, a detailed equipment usage timeline was developed for the baseline mission and is given in Table 4.38. Required G and N system performance during the various mission phases was formulated from the duty cycle estimates and functional flow logic of the baseline mission. Timeline details estimated were then based on accuracy and system operational requirements. Constraints and ground rules derived from Apollo experience and adhered to are as follows:

- The G and N system performs all thrust, precise attitude, and entry controls following transplanet injection. The Attitude and Stability Control System (A and SCS) provides attitude stabilization during periods when lesser attitude accuracy is acceptable.
- The G and N system performs a monitoring function from liftoff through transplanet injection.
- The G and N system performs star-landmark navigation measurements during the planet encounter phase.
- No G and N functions are required during the artificial gravity phases.
- After shutdown, a warmup period of one hour is required before aligning the inertial reference. However, no warmup of the optics or computer equipment is necessary.
- A complete alignment of the stable member is assumed to take 15 minutes. Where time is critical, the alignments are scheduled to be completed at least 15 minutes prior to needing the inertial reference.
- Stable member alignment updates are assumed to occur every 1.5 hours during the planet encounter phase, and every 4 hours during the earth orbit, planet approach, and earth approach phases.
- A star-landmark angle measurement via the sextant (SXT) is assumed to take 6 minutes.
- An average of one navigation measurement every two hours is assumed during the planet encounter phase.
- Usage times for the G and N system reliability calculations include the prelaunch alignment time.

Mission Phase	Duration (Hours)	G&N Functions	Equipment Usage (Hours)		
			ISS	OSS	CSS
Prelaunch	96	Vert. Erection, Optical Az. Align., and Gyrocompassing Alignment	4.0	0.10	4.0
Boost-First Stage		Monitor Attitude and ΔV			
Boost-Second Stage and Injection	24	Monitor Attitude and ΔV	24.0	0.42	24.0
Waiting Orbit Coast		Monitor Att. and 5 Align. Updates			
Inter-Orb.Man. and P. O. Injection		Monitor Attitude and ΔV			
Earth Orbit	48	Monitor Att., 1 Align., and 5 Align. Updates	24.0	0.67	24.0
Transplanet Injection	0.7	Monitor Attitude and ΔV and 1 Align. Update	0.7	0.08	0.7
Transplanet Coast					
Zero-g Coast	168	1 Alignment, Attitude Control, and ΔV Control	2.0	0.25	2.0
Spin	1518.1	None	0	0	0
Midcourse Correction	168	1 Alignment, Attitude Control, and ΔV Control	2.0	0.25	2.0
Spin	1516.9	None	0	0	0
Planet Approach	20.0	1 Alignment, 4 Align. Updates, Att. Control, and ΔV Control	18.0	0.58	18.0
Planet Encounter	460	1 Alignment, 266 Align. Updates, Att. Control, and 230 Star-Landmark Navigation Sightings	400.0	45.41	400.0
Transearth Coast					
Zero-g Coast	168	1 Alignment, Attitude Control, and ΔV Control	2.0	0.25	2.0
Spin	6187.6	None	0	0	0
Midcourse Correction	168.0	1 Alignment, Attitude Control, and ΔV Control	2.0	0.25	2.0
Spin	6186.4	None	0	0	0
Earth Approach	20.0	1 Alignment, 4 Align. Updates, Att. Control, and ΔV Control	18.0	0.58	18.0
Earth Retro	0.5	Att. Control and ΔV Control	0.5	0	0.5
Earth Entry	0.36	Entry Control	0.36	0	0.36
Recovery	72	None	0	0	0
Total	16,822.56		497.56	48.84	497.56

Table 4.3.3 Timeline - Mars Flyby Mission

- It was assumed that the first transplanet velocity correction (ΔV) could occur anywhere within the zero-g coast period, but not necessarily at the beginning. Therefore, an additional warmup period and IMU alignment were required which would not occur if the platform remained on following injection.
- System status monitoring is performed whenever the G and N system is operative.

The timeline indicates that the total inertial and computer subsystem (CSS) usage times are approximately 500 hours, and the optical subsystem (OSS) usage time is approximately 50 hours for the indicated mission. Computer subsystem usage time is no greater than the inertial subsystem (ISS) time since optics functions are planned only when the inertial subsystem is operative. Therefore, the computer never has to operate separately with the optics subsystem.

Usage times are broken down into major subsystem usage times and not G and N functional times. This is possible because the various G and N functions, in general, use the same equipment. For example, the same inertial subsystem equipment is energized when performing attitude control, entry control, ΔV control, monitor, warmup, alignment, checkout and status monitor. The same optical subsystem equipment is energized when performing optical alignment, navigation, checkout, and status monitor. The entire CSS is used whenever the ISS or OSS is used. The one exception to exciting all equipment in a subsystem when any one part of the subsystem is necessary, occurs during the earth retro burn when the digital to analog converters of the optics coupling data units are used to relay the main engine gimbal signals.

Equipment failure rates are derived from Reference 4.15 and apply to Apollo Block II equipment. The major subsystem and subassembly failure rates are given in Table 4.34.

Table 4.34 G and N Subsystem and Function Failure Rates

Subsystem	$\lambda \times 10^{-6}$	Subassembly	$\lambda \times 10^{-6}$
Inertial (ISS)	394	IMU and PIPA Elec. Assy	129
		PSA	110
		CDU	155
Optical (OSS)	264.3	Optical Assy.	94
		PSA	77
		CDU	91
		D&C Group	2.3
Computer (CSS)	237.3	AGC	235
		DKSY	2.3

Not all the display and monitoring equipment is included, even though many G and N system signals might be displayed there. Only the DSKY's and the Displays-and Controls group (D and C) are considered G and N equipment. The DSKY's are normally redundant, accounting for the low failure rate, and the D and C group is considered part of the optical subsystem since it is used only during optics functions.

Applying the subsystem usage times to the subsystem failure rates as shown in Table 4.35 it is seen that the reliability for a single G and N system in the defined mission is 0.7195. A redundant unit for each of these subsystems now yields a reliability of 0.9761.

The optical subsystem practically meets the overall G and N system reliability goal while the inertial and computer subsystems are considerably lower. Improvement is obviously required in all areas. If the operational concept is not changed and the usage times remain the same; the reliability is inadequate for the proposed mission.

4.8.3 Availability Analysis - G and N System

The next step in isolating the weak reliability links, to enable determination of the optimum means of improving the system contribution to the probability of safe Return (P_s), is shown in the subassembly reliability breakdown of Table 4.36 , Here, it is shown that the three major subassembly reliabilities for the inertial subsystem are of comparable magnitude and all need improvement. The optical subsystem failures are of an order of magnitude less than those of the inertial subsystem, but some improvement is still necessary in some of these if the overall Gand N system reliability goal is to be met. The Display and Controls Group (D and C) is obviously good enough. The computer subsystem is broken down into the computer itself and the normally redundant display and keyboard (DSKY). Only the computer itself needs improvement.

Examining the subassembly level, it is clear that a total of eight spares (approximately 350 pounds) are required to bring the total G and N system P_s to better than 0.9920. A smaller amount of spare equipment than that shown would actually be needed for sparing at a lower assembly level. For a given amount of additional equipment or weight, normally, greater gains in reliability can be achieved at the lower level of the spared item.

For example, if the AGC failure rate could be divided into four equal portions representing four separate assemblies and a spare for each assembly was available, rather than just a complete spare computer, the overall G and N reliability goal would be met. By sparing at the whole computer level, two spare computers are required. Similar arguments can

Table 4.35. Guidance and Navigation Subsystem Level Availability Analysis

Subsystem	$\lambda \times 10^{-6}$	T HRS	λT	Reliability	No. of Spares	Equivalent λT	Contribution to P_S
ISS	394	500	0.197000	0.8210	1	0.017420	0.9827
OSS	264.3	50	0.013215	0.9869	1	0.000109	0.9999
CSS	237.3	500	0.118650	0.8880	1	0.006640	0.9934
Total			0.328865	0.7195	3	0.024170	0.9761

Table 4.36. Functional Level Availability Analysis, G and N System Mars Flyby Mission

Item	$\lambda \times 10^{-6}$	T HRS	λT	Reliability	No. of Spares	Equivalent λT	Contribution to P_S
IMU and PIPA Elec. Assy.	129.0	500	0.064500	0.9375	1	0.0020	0.9980
PSA (ISS)	110.0	500	0.055000	0.9465	1	0.0014	0.9986
CDU (ISS)	155.0	500	0.077500	0.9254	1	0.0029	0.9971
AGC	235.0	500	0.117500	0.8890	2	0.00041	0.9996
DSKY	2.3	500	0.001150	0.9989	0	0.001150	0.9989
Optical Assembly	94.0	50	0.004700	0.9953	1	0.0000221	0.9999
PSA (OSS)	77.0	50	0.003850	0.9962	1	0.0000148	0.9999
CDU (OSS)	91.0	50	0.004550	0.9955	1	0.0000207	0.9999
D&C	2.3	50	0.000115	0.9999	0	0.000115	0.9999
Total			0.328865	0.7195	8	0.008033	0.9920

be applied to the other assemblies with a resultant decrease in total weight and volume of spares required, but an increase in malfunction isolation difficulty and maintenance.

Even at the subassembly level some juggling of assemblies and the associated spares required is possible. If, instead of the spares listed in Table 4.36 two spares of the IMU and PIPA electronics assembly and the ISS portion of the CDU were provided, only one spare computer would be needed. Also, if minor improvements were made, no spare of the optical assembly would be required. This would be desirable because portions of the optics assembly are outside the spacecraft. Only the electronics part of the optical assembly would have to be spared in any event since the glass is completely reliable when protected.

Approximately 80 percent of the ISS and CSS usage times and over 90 percent of the OSS usage time occurs during the planet encounter phase due to the local vertical hold and navigation requirements. Therefore it would be well to re-evaluate the necessity of these requirements for the total time durations. Impact on redesign and spares/redundancy will be great if these planet encounter usage times vary significantly. As a counter example, assume on-board navigation is only backup to ground tracking and is not required behind the planet. Also assume that an accurate alignment and attitude hold is only required for an hour early and an hour late in planet encounter when probes are launched and collected. ISS and CSS usage becomes approximately 100 hours and CSS time becomes approximately 4 hours. For this mission, in which the G and N system performs boost monitor, ΔV control, short-term attitude control, and entry control, the present Apollo G and N system will meet the reliability constraints with no modifications (except turning equipment OFF during non-operational periods) as shown in Tables 4.37 and 4.38.

The inertial subsystem subassemblies and the computer reliabilities are slightly low for this mission application while the optical subsystem subassemblies are much better than needed. Considering spares for those subassemblies with reliabilities less than 0.99, it is seen that four spares (approximately 175 pounds) are required. However, the resultant failure total in the mission is more than five times better than is necessary. The same arguments concerning sparing at a lower equipment level than the subassembly level as set forth for the other timeline are even more applicable in this situation.

Spares below the subsystem component level was not considered for the optical subsystem since they were not required. They were also not considered for the inertial subsystem below the component level because of the impracticality of maintenance or repair.

Table 4.37. Guidance and Navigation Subsystem Level Analysis, Reduced Duty Cycle Mode

Subsystem	$\lambda \times 10^{-6}$	T HRS.	λT	P_S	No. of Spares	Equivalent λT	Equivalent P_S
ISS	394	100	0.039400	0.9613	1	0.000825	0.9992
OSS	264.3	4	0.001057	0.9989	0	0.001057	0.9989
CSS	237.3	100	0.023730	0.9766	1	0.000225	0.9998
Total			0.064187	0.9378	2	0.002107	0.9979

Table 4.38. Functional Level Availability Analysis, G and N System, Reduced Duty Cycle Mode

Item	$\lambda \times 10^{-6}$	T Hours	λT	P_S	No. of Spares	Equivalent λT	Equivalent P_S
IMU & PIPA Elec. Assy.	129.0	100	0.012900	0.9872	1	0.000065	0.9999
PSA (ISS)	110.0	100	0.011000	0.9890	1	0.000121	0.9999
CDU (ISS)	155.0	100	0.015500	0.9846	1	0.000139	0.9999
AGC	235.0*	100	0.023500	0.9768	1	0.000245	0.9998
DSKY	2.3	100	0.000230	0.9998	0	0.000230	0.9998
Optical Assembly	94.0	4	0.000376	0.9996	0	0.000376	0.9996
PSA (OSS)	77.0	4	0.000308	0.9997	0	0.000308	0.9997
CDU (OSS)	91.0	4	0.000364	0.9996	0	0.000364	0.9996
D&C	2.3	4	0.000009	0.9999	0	0.000009	0.9999
Total			0.064187	0.9378	4	0.001857	0.9981

* Based on MIT-predicted value, Reference 4.8.2

Up to this point, the performance of one G and N system with spares has been considered in two different mission applications. In actuality, two complete G and N systems, one in the Earth Entry Module (EEM) and one in the mission module (MM) are planned to provide the required mission functions. No reliability is gained if only one of the systems is available at any one time. On the other hand, little or no additional spare equipment would be necessary if the two sets of equipment were always available to back up one another. The actual situation will most likely lie somewhere between these extremes.

Some factors to be considered are as follows:

- The astronauts will be in the EEM at least during boost, major thrusts, and entry. Therefore, for monitoring purposes, at least some of the EEM G and N equipment must be operating, while all the EEM G and N equipment must be operative following EEM/MM separation prior to entry.

- The EEM optics head is covered by a shroud whenever the EEM and MM are docked. Therefore, the only time the EEM is uncovered and astronauts are in the EEM is when an optical system is needed late in the mission after EEM/MM separation prior to entry.

- During the artificial gravity modes, the vehicles are no longer docked, but all the astronauts are in the MM. Therefore, the entire EEM G and N system is unavailable for use unless some data coupling or switching mechanism is devised.

- Accurate alignment of the EEM inertial reference with the MM optics is probably not possible because of large misalignments between the two mounting bases. Possibly, the EEM and MM airframes could be aligned accurately before launch, or at least the misalignment could be calibrated, but this accuracy would be lost once the vehicles had separated.

It would appear that some cross switching between the EEM and MM computer subsystems or their subassemblies will be desirable since EEM display equipment is needed during the monitoring phases when the inertial reference is probably in the MM. Also, it would be desirable to have the attitude and navigation data placed into the EEM computer prior to EEM/MM separation before entry even though it was obtained using the MM optics.

4.8.4 Availability Analysis - Guidance Computer (AGC)

Note: Some of the material used herein was provided by the Raytheon Company through References 4.16 and 4.17.

The AGC presents the highest failure hazard of any of the G and N subsystems. For this reason and because of its modular design, so amenable to maintenance, it is desirable to explore the effects of sparing at the lower levels of assembly.

Table 4.39 presents a breakdown of the AGC modules along with the projected failure rates for both operating and standby states. There are a total of 48 modules in the AGC, 38 are different types and all of them must operate for the 497.6 hours of Table 4.33. The resultant mean time before failure (MTBF) is 2878 hours and the probability of no failure or $P_s = 0.74$ for the mission and 0.84 for the active time only; this is, of course, too low. Also, it does not tell the whole story; about 1.85 hours of this time is critical since little or no time exists for maintenance or repair and therefore, the downtime constraint may not be satisfied. However, the assessed MTBF assures a 0.99936 chance of no failure during the critical phases of the mission provided it is functioning at the start. This is well above the requirement. Therefore, the only periods where the probability of a failure is unsatisfactory is associated with the zero-gravity mission phases where no commitments are scheduled and maintenance is possible.

From Table 4.36 it is evident that there are several weaker links in the AGC modules and the remainder are fairly equal. Some of the modules are interchangeable which will permit canabilization and a reduced number of supporting spares.

Figure 4.55 presents the results of various sparing concepts as applied to the AGC for various duty cycles. It indicates that it would take two spare computers, or about 38 spare modules to provide a P_s in excess of 0.999. Canabalizations would improve the results of using two completely spared AGC's. However, the associated weight penalty of 112 pounds is excessive since the modules required for the same P_s would be about one-sixth of that weight.

Conducting a spares requirements analysis at the module level in an attempt to reduce the risk of no correctable failure to less than 0.999 provided the results recorded in Table 4.37, each module function must exceed 0.999973 to achieve the goal. By providing the 41 spares indicated therein and with no changes made to the AGC, the P_s was reduced to less than 0.99975 and thereby assures meeting the mission requirement safely. The resulting weight requirements summary is presented in Table 4.41 along

Table 4.39 Breakdown of Computer Module

Module Name	No. of each Type (N)	Operating Failure Rate/ Module (λo)	Standby Failure Rate/ Module (λs)	N λo	Nλs
A1 Scalar	1	6.6625	2.0225	6.6625	2.0225
A2 Timer	1	6.8825	2.0825	6.8825	2.0825
A3 Sequence Register	1	6.6625	2.0225	6.6625	2.0225
A4 Stage Branch	1	6.7725	2.0525	6.7725	2.0525
A5 Cross Point Gen. I	1	6.3875	1.9475	6.3875	1.9475
A6 Cross Point Gen. II	1	6.0575	1.8575	6.0575	1.8575
A7 Service Gate	1	6.8825	2.0825	6.8825	2.0825
A8 Through All Four Bit	4	6.8825	2.0825	27.5300	8.3300
A12 Parity	1	6.6075	2.0075	6.6075	2.0075
A13 Alarm	1	6.7175	2.0375	6.7175	2.0375
A14 Memory Timing	1	6.7175	2.0375	6.7175	2.0375
A15 Rupt Service	1	6.6625	2.0225	6.6625	2.0225
A16 IN OUT I	1	6.8825	2.0825	6.8825	2.0825
A17 IN OUT II	1	6.8825	2.0825	6.8825	2.0825
A18 IN OUT III	1	6.8825	2.0825	6.8825	2.0825
A19 IN OUT IV	1	6.8275	2.0675	6.8275	2.0675
A20 Counter Cell I	1	6.8825	2.0825	6.8825	2.0825
A21 Counter Cell II	1	6.7725	2.0525	6.7725	2.0525
A22 IN OUT V	1	6.7175	2.0375	6.7175	2.0375
A23 IN OUT VI	1	6.7175	2.0375	6.7175	2.0375
A24 IN OUT VII	1	6.7175	2.0375	6.7175	2.0375
A25, A26 Interface	2	7.3920	0.7561	14.7840	1.5122
A27-A29 Interface	3	9.9450	0.7751	29.8350	2.3253
A30, A31 Power Supply	2	6.2765	0.5958	12.5530	1.1896
B7 Clock Oscillator	1	2.2845	0.2267	2.2845	0.2267
B8 Alarm	1	5.3470	0.8213	5.3470	0.8213
B9, B10 Erasable Driver	2	7.0200	1.0129	14.0400	2.0258
B11 Current Switch	1	3.2170	0.4450	3.2170	0.4450
B12 Erasable Memory	1	5.4533	3.7253	5.4533	3.7253
B13, B14 Sense Amplifier	2	15.5530	2.3364	31.1060	4.6728
B15 Strand Select	1	4.9565	0.8255	4.9565	0.8255
B16, B17 Rope Driver	2	6.4340	0.8665	12.8680	1.7330
B21 Rope Memory 1	1	8.2080	1.5968	8.2080	1.5968
B22 Rope Memory 2	1	8.2080	1.5968	8.2080	1.5968
B23 Rope Memory 3	1	8.2080	1.5968	8.2080	1.5968
B24 Rope Memory 4	1	8.2080	1.5968	8.2080	1.5968
B25 Rope Memory 5	1	8.2080	1.5968	8.2080	1.5968
B26 Rope Memory 6	1	8.2080	1.5968	8.2080	1.5968
Totals	48			347.7173	78.1483
			MTBF x 2878 Hours		

Table 4.40. Sparing Analysis, Guidance Computer, Module Level

Item	NλT	R.	w/one Spare	w/two Spares
A1	0.00331526	0.996685	0.9999946	
A2	0.00342473	0.996575	0.9999942	
A3	0.00331526	0.996685	0.9999946	
A4	0.00336999	0.994430	0.9999942	
A5	0.00317842	0.996822	0.9999950	
A6	0.0030142	0.996986	0.9999955	
A7	0.0034247	0.996575	0.9999942	
A8-11	0.0136989	0.996301	0.9999050	0.9999995
A12	0.0032879	0.996712	0.9999946	
A13	0.00334263	0.996657	0.9999946	
A14	0.00334263	0.996657	0.9999946	
A15	0.00331526	0.996685	0.9999946	
A16	0.00342473	0.996575	0.9999942	
A17	0.00342473	0.996575	0.9999942	
A18	0.00342473	0.996575	0.9999942	
A19	0.00339736	0.996603	0.9999942	
A20	0.00342473	0.996575	0.9999942	
A21	0.00336999	0.996630	0.9999942	
A22	0.0033426	0.996657	0.9999946	
A23	0.0033426	0.996657	0.9999946	
A24	0.0033426	0.996657	0.9999946	
A25-26	0.007358	0.992642	0.9999720	
A27-29	0.014459	0.985154	0.99988	0.9999994
A30-31	0.0062463	0.993754	0.9998810	
B7	0.00113677	0.998863	0.9998994	
B8	0.0026606	0.997339	0.9998965	
B9-10	0.0069863	0.993914	0.9998750	
B11	0.00160077	0.998399	0.9998987	
B12	0.0027135	0.997286	0.9998967	
B13-14	0.0154783	0.984522	0.9987	0.9999994
B15	0.00246635	0.997534	0.9987971	
B16-17	0.00640311	0.993597	0.9987978	
B21	0.0040843	0.995916	0.9987918	
B22	0.0040843	0.995916	0.9987918	
B23	0.0040843	0.995916	0.9987918	
B24	0.0040843	0.995916	0.9987918	
B25	0.0040843	0.995916	0.9987918	
B26	0.0040843	0.995916	0.9999918	
	Totals	0.84	0.99941	0.99975

Table 4.41 . Spares and Weight Requirements

Module - Name	Identification Number	Module Wt. in Grams	No. Req.	Total Wt. in Grams
Scaler	A 1	163	1	163
Timer	A 2	163	1	163
Sequence Register	A 3	163	1	163
Stage Branch	A 4	163	1	163
Cross Point Gen. I	A 5	163	1	163
Cross Point Gen. II	A 6	163	1	163
Service Gate	A 7	163	1	163
Four Bit	A 8 to A11	163	2	326
Parity	A12	163	1	163
Alarm	A13	163	1	163
Memory Timing	A14	163	1	163
Rupt Service	A15	163	1	163
IN OUT I	A16	163	1	163
IN OUT II	A17	163	1	163
IN OUT III	A18	163	1	163
IN OUT IV	A19	163	1	163
Counter Cell I	A20	163	1	163
Counter Cell II	A21	163	1	163
IN OUT V	A22	163	1	163
IN OUT VI	A23	163	1	163
IN OUT VII	A24	163	1	163
Interface	A25 & A26	165	1	165
Interface	A27 to A29	200	2	400
Power Supply	A30 & A31	508	1	508
Clock Oscillator	B 7	180	1	180
Alarm	B 8	310	1	310
Erasable Driver	B 9 & B10	250	1	250
Current Switch	B11	297	1	297
Erasable Memory	B12	272	1	272
Sense Amplifier	B13 & B14	202	2	202
Strand Select	B15	316	1	316
Rope Driver	B16 & B17	297	1	297
Rope Memory 1	B21	904	1	904
Rope Memory 2	B22	904	1	904
Rope Memory 3	B23	904	1	904
Rope Memory 4	B24	904	1	904
Rope Memory 5	B25	904	1	904
Rope Memory 6	B26	904	1	904
		Totals Required	41	12,207*

*Equals 26.91 pounds

with the total number of individual spares recommended, by module. Again, these estimates do not reflect the effects of module repair and interchangeability of EEM and MM modules which would further improve the picture.

4.8.5 G and N Maintainability

Present design constraints limits the ability to maintain or repair most of the inertial subsystem components; and the optical subsystem is not expected to require either.

The AGC is the major contributor to the failure hazard and is repairable in its present state. However, it could be improved, in terms of reducing the repair time by changing the method of fastening the container so as to decrease the access time. Refer to Figure 4.56 where the Apollo II AGC is depicted, notice that there are about 40 screws to remove to gain access to either module tray and 18 more to remove it from the mounting. Once access is made, it is very easy to replace any of the modules as shown in Figures 4.57 and 4.58. Even the logic modules themselves display some potential repairability as seen from Figure 4.59.

The conclusions to be drawn relevent to the maintenance and repair potential of the G and N system is that it is maintainable where needed most. Those aspects not maintainable are either not required, or it is more practical to replace them at a component level. For an example of the latter, see the Display and Keyboard (DSKY) of Figure 4.60.

4.8.6 Summary and Conclusions

This analysis establishes the feasibility of using the Apollo Guidance and Navigation (G and N) equipment in long-term missions, as exemplified by the baseline Mars Flyby mission. It does however, identify areas where further study would be needed to arrive at an optimum system configuration for any defined mission application.

The results would indicate a reasonable amount of spare equipment is necessary, depending on mission, system application, and reliability improvement concept followed. The recommended operational concept is as follows:

1. G and N equipment must be turned completely OFF during the non-controlling standby periods of the mission, that is, during the artificial gravity phases.
2. On-board maintenance must be performed using spare equipment provided with some provisions for scavenging between the mission module system and the EEM just prior to reentry. A total of

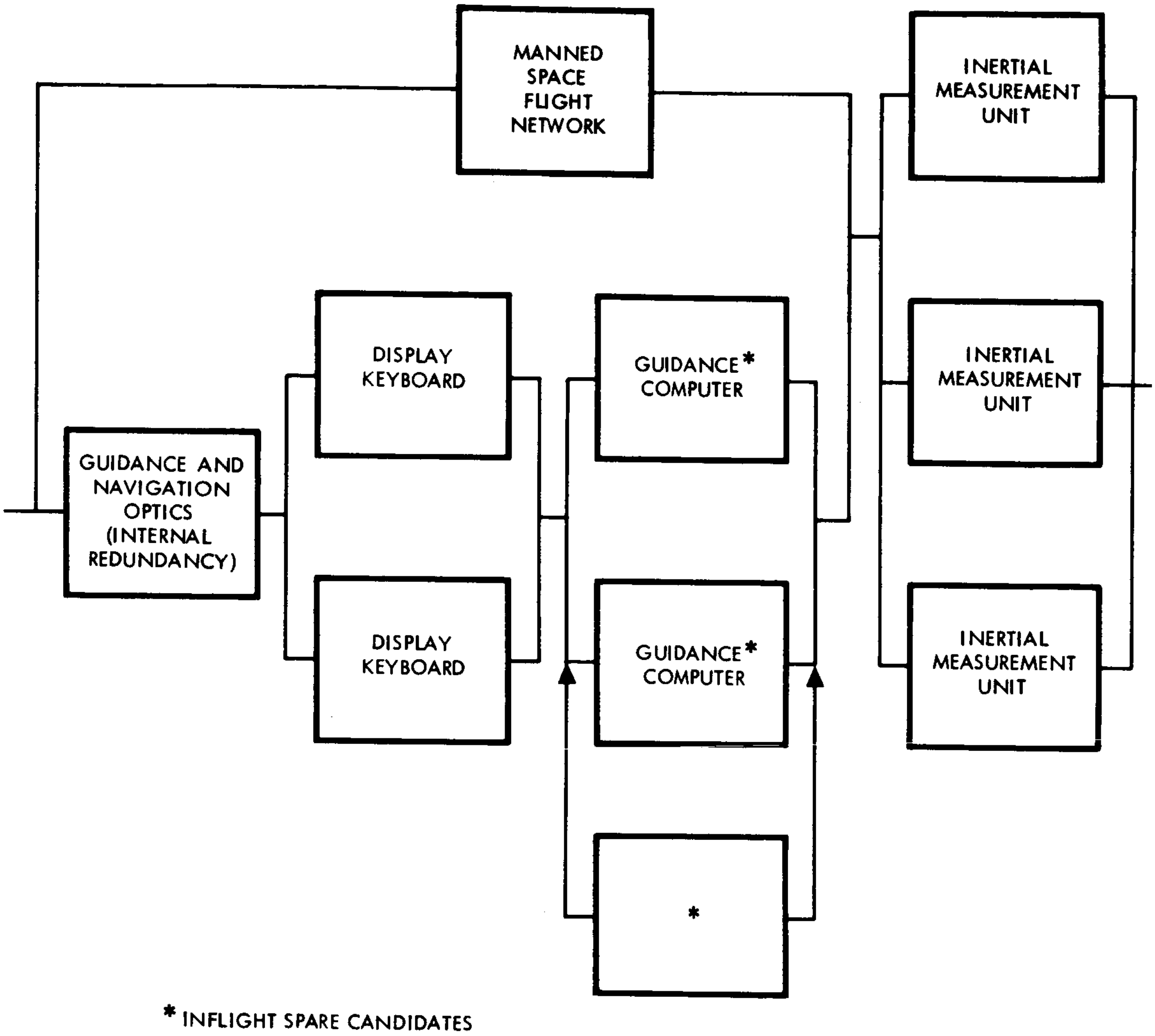


Figure 4.57. Electronic Subsystem

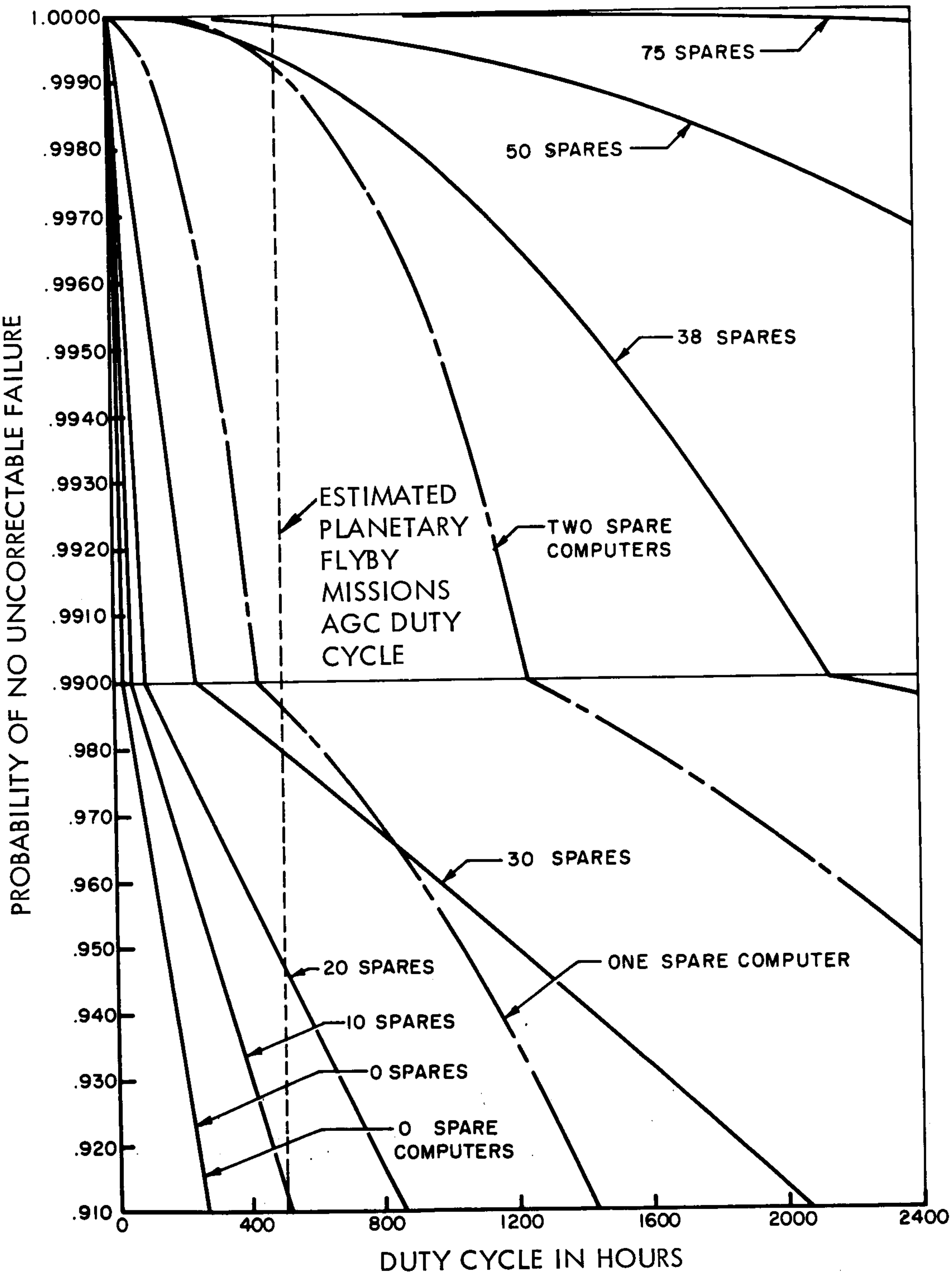


Figure 4.58. AGC Contribution to Probability Safe Return Vs Time for Various Sparing Philosophies

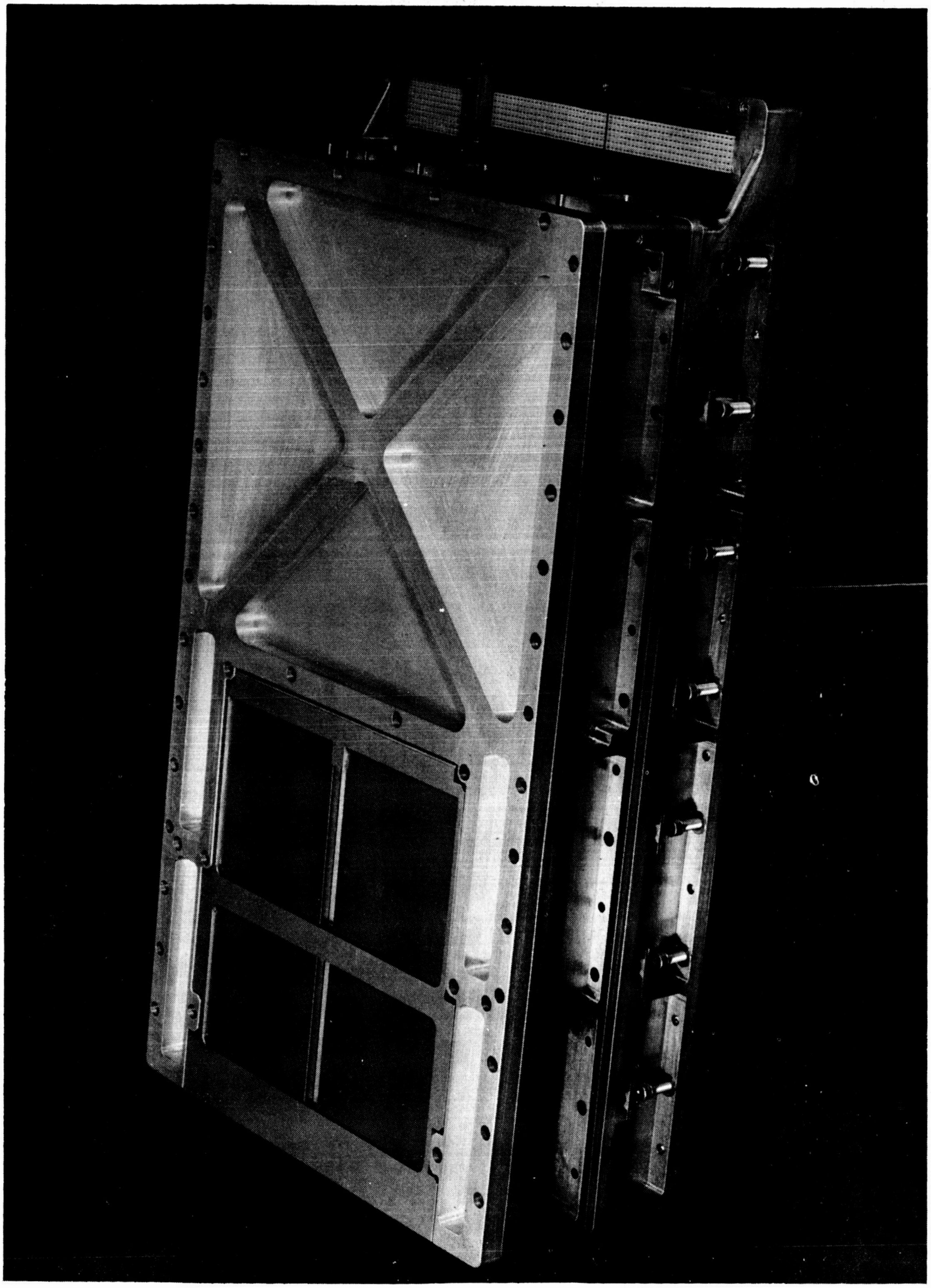


Figure 4.59. Block II AGC, External View

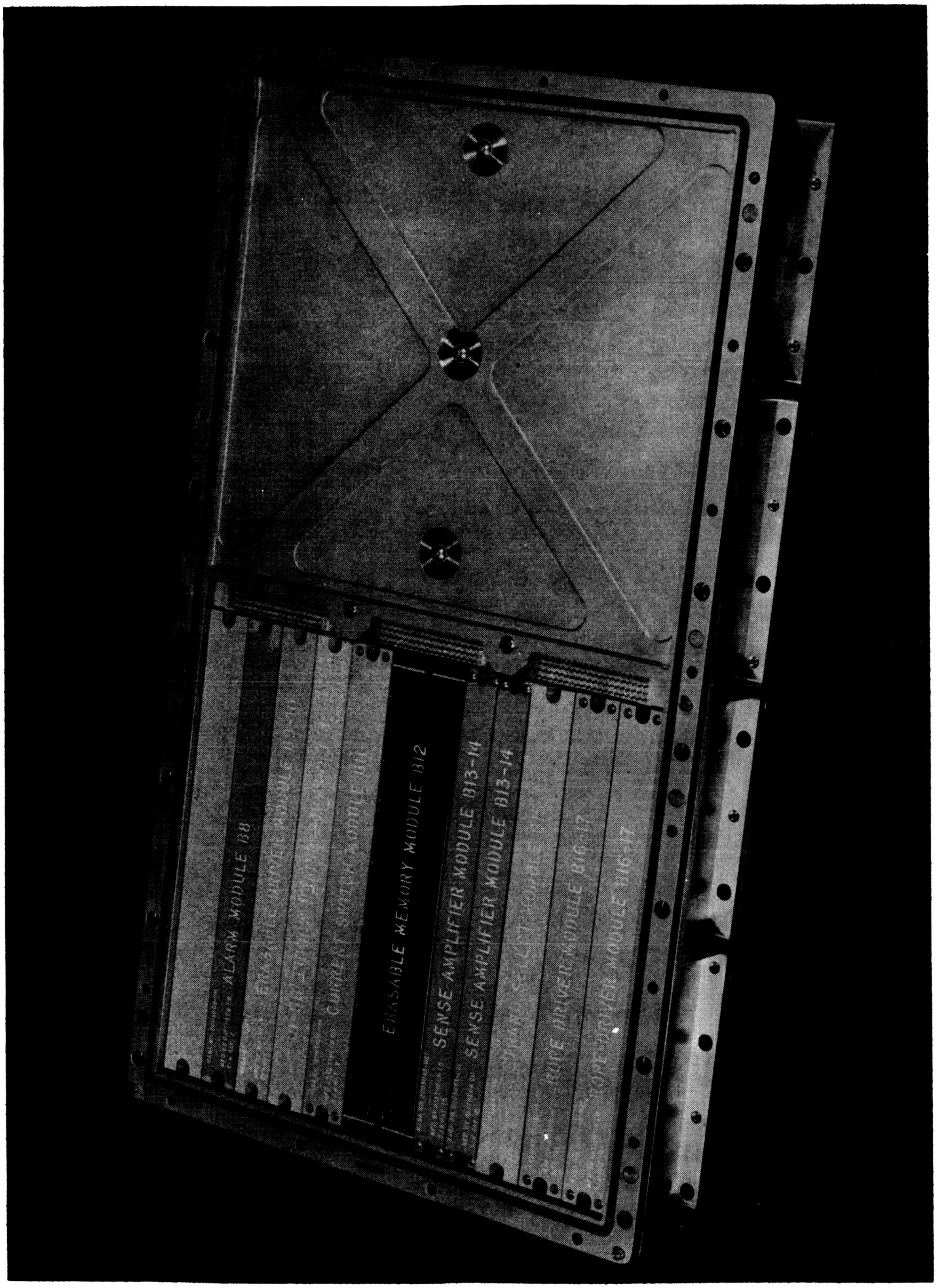


Figure 4.60. Block II AGC Tray B

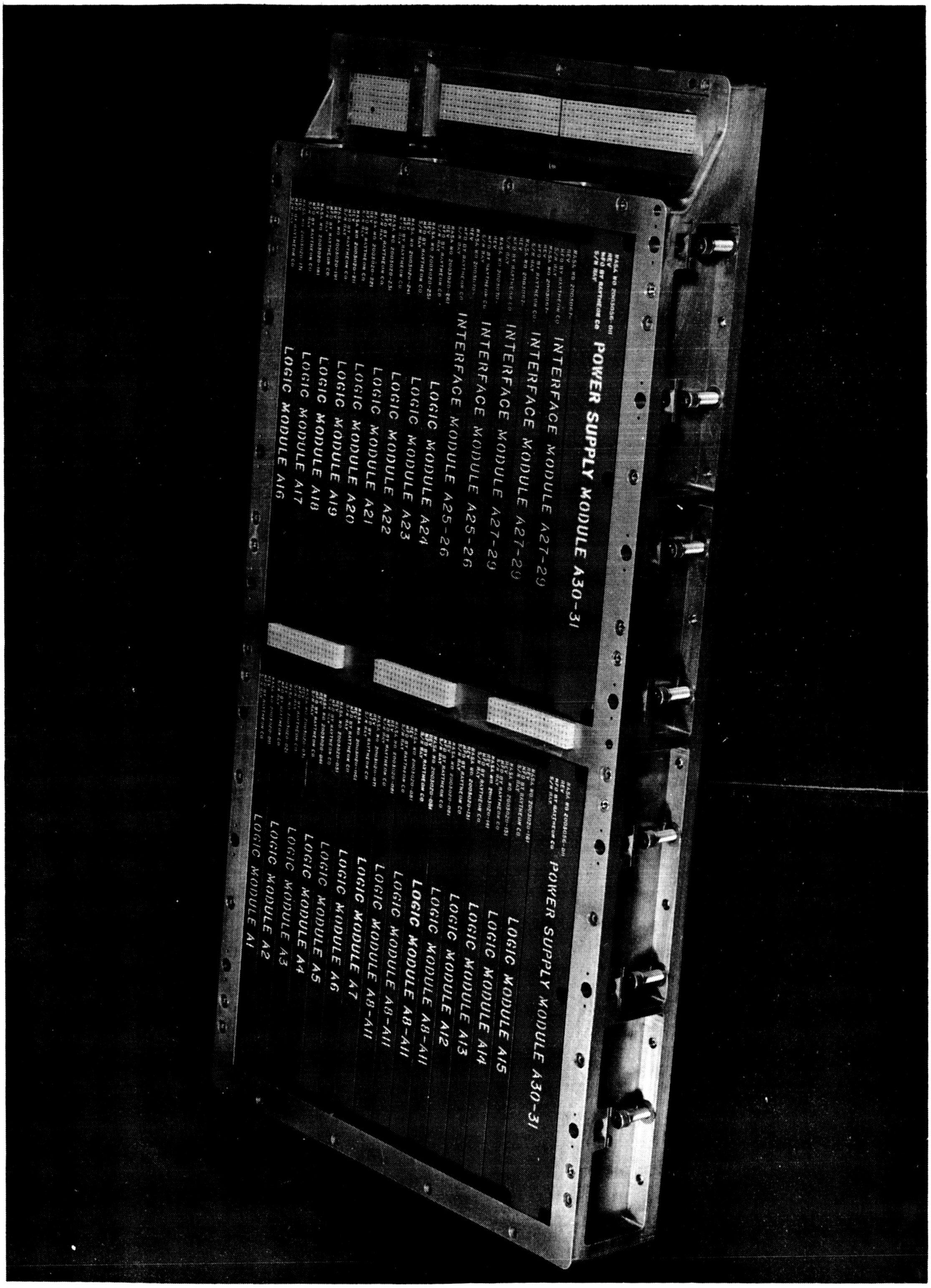


Figure 4.61. Block II AGC Tray A

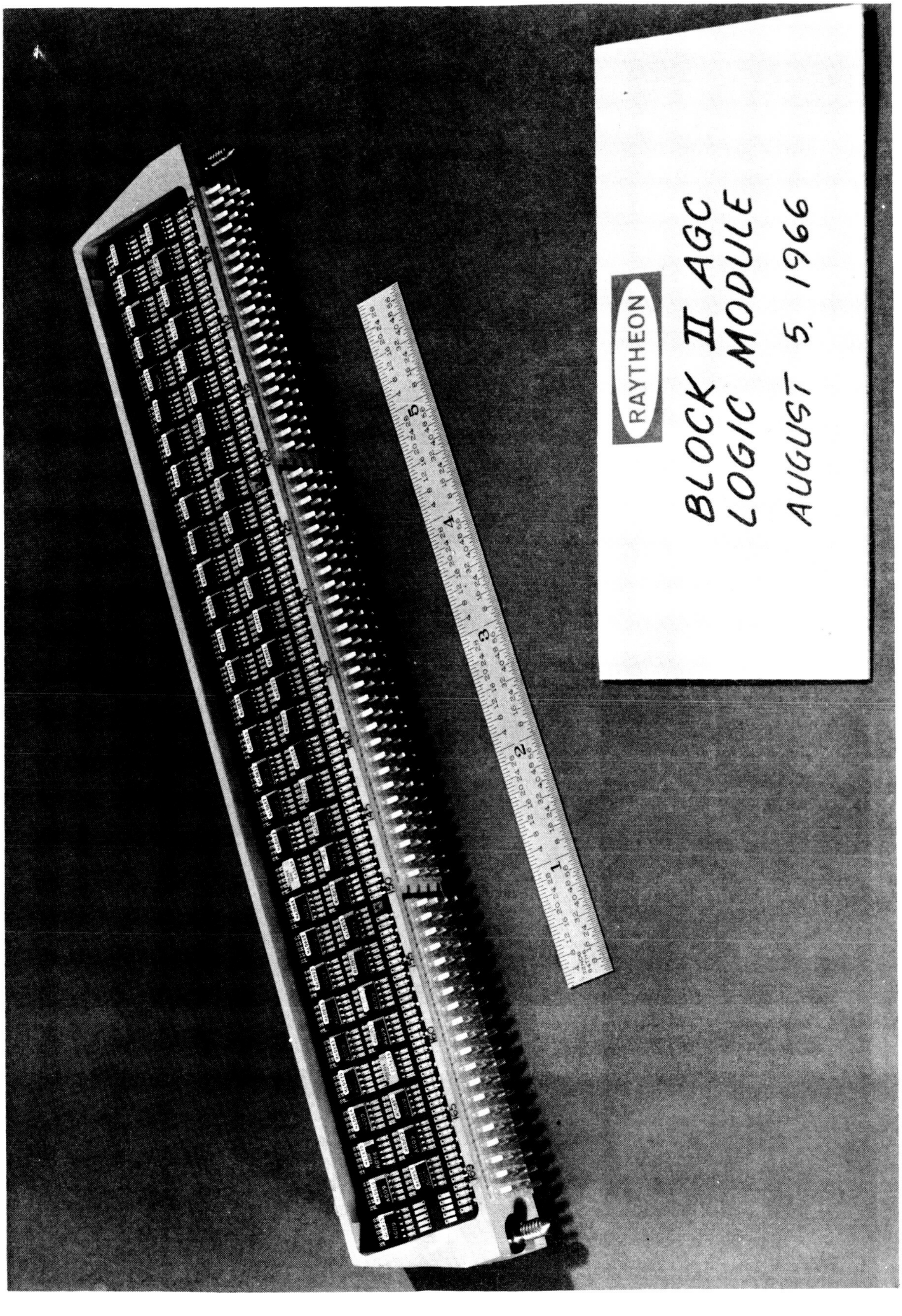


Figure 4.62. Block II AGC Logic Module

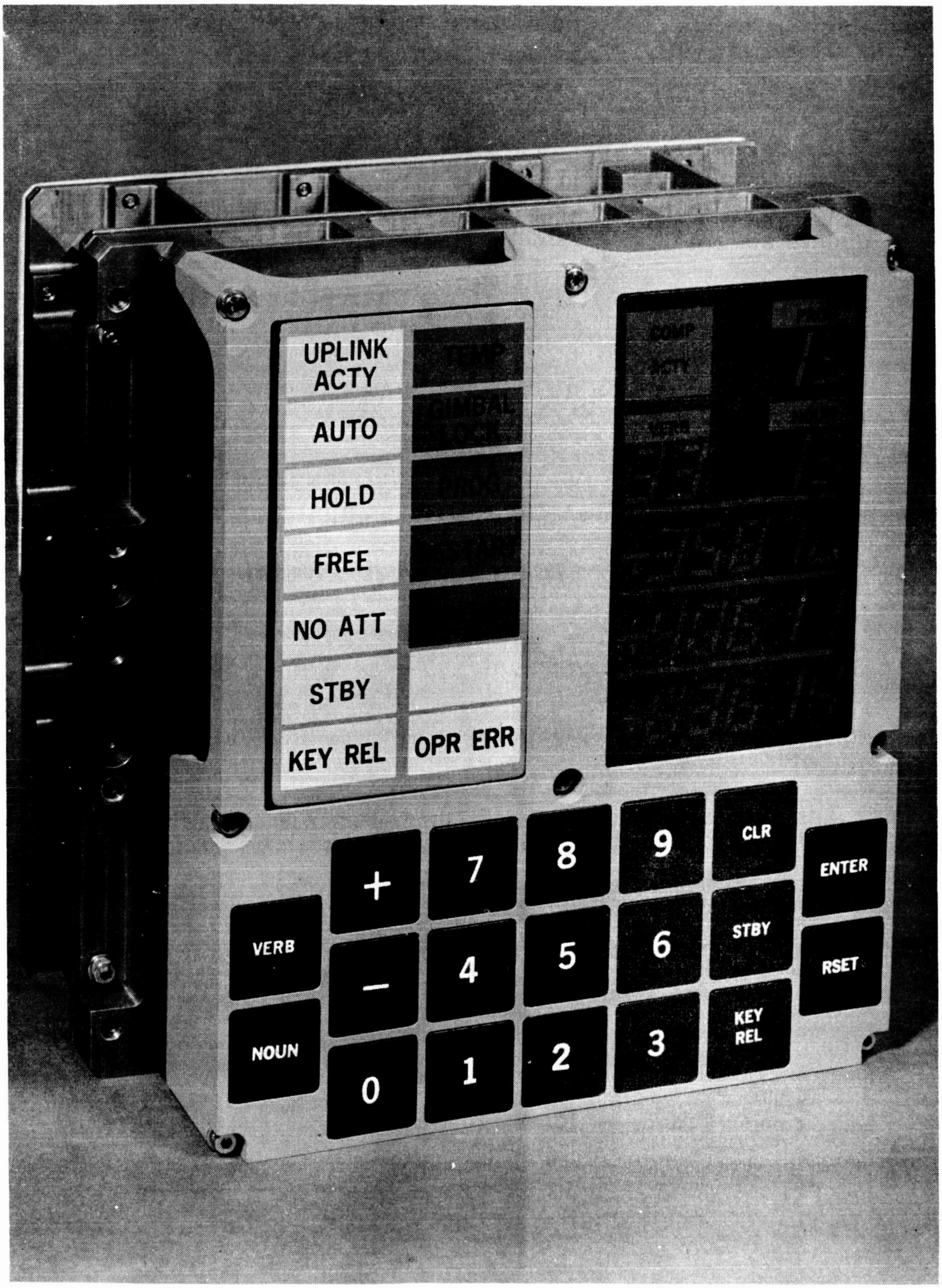


Figure 4. 63. Display and Keyboard

about 3 spares at the functional assembly level are required for the ISS and OSS. Forty-one spares are required for the CSS at the module level.

Areas where further effort is necessary are as follows:

1. Determination of the available and the required computer time update accuracy after shutdown.

2. Investigation of inertial instrument performance characteristics after cooldown and lack of suspension during shutdown.

3. Determine maintainability characteristics of stable platforms (IMU).

4. Detailed analysis of the precise G and N system functional requirements for development of required equipment usage and duty cycles.

5. Determination of the optimum of weak-link reliability improvement based on redesign possibilities, ease of maintenance, malfunction detection and isolation, and weight and volume penalty of recommended additional equipment.

6. Determination of mounting and packaging changes required for maintenance.

7. Consideration of means of coupling data and cross switching of components between the two on-board systems and/or the EEM/MM.

4.9 ELECTRICAL POWER SYSTEM

The electrical power system (EPS) is composed of three subsystems, the main power source, the power distribution and processing system and the batteries. Each of these have been examined in detail to determine the implications of extended missions on the subsystems design and support requirements.

The functions of the electrical power system are presented in Figure 4.64. It must provide the required energy to operate the spacecraft systems. Total loss of the function for an extended period would be catastrophic to the missions and the crew. Temporary loss is permissible for a short period of time. (See Volume I on downtime constraints.) The EPS is a Criticality I system, and therefore, must not create any failure hazard that cannot be maintained within the accepted risk level; that is, about 0.999.

4.9.1 Electrical Power Source

Note: Much of the data used herein was derived from a study conducted by NAR/AI in Reference 4.18 and NAR/SD in the baseline mission study, Reference 1.1.

Functional Requirements and Selected System Concept

The baseline mission study recommended two concepts for the electrical power source, a solar voltaic system for a zero-gravity mission and an isotopic system for the selected baseline mission. Since this study recommends the artificial gravity mode for the major portion of the mission, the isotopic system is recommended and was used herein as the baseline system. The solar voltaic is very unreliable and not amenable to M&R or reliable operation in the artificial gravity environment.

Of those radioisotope systems considered, the Rankine Dowtherm A conversion system was selected as the most likely isotope system which could be developed in time for a 1975 mission. This is based on its low temperature requirement and the minimum difficulty anticipated in its development all components are within the state of the art.

A power level of 7.0 kWe was established for a four-man crew as average power required for the peak loads during the 20 days of planet encounter. Other mission periods will require an average of 6.0 kWe. This provided the basis of sizing the isotope systems. An exception to this requirement was made for the case of solar photovoltaic. It was determined that for the Mars Twilight mission the array size could be set at a power output of 5.5 kWe and the array sized at the farthest solar distance of 2.2 A.U. This provided approximately 7.8 kWe at Mars encounter.

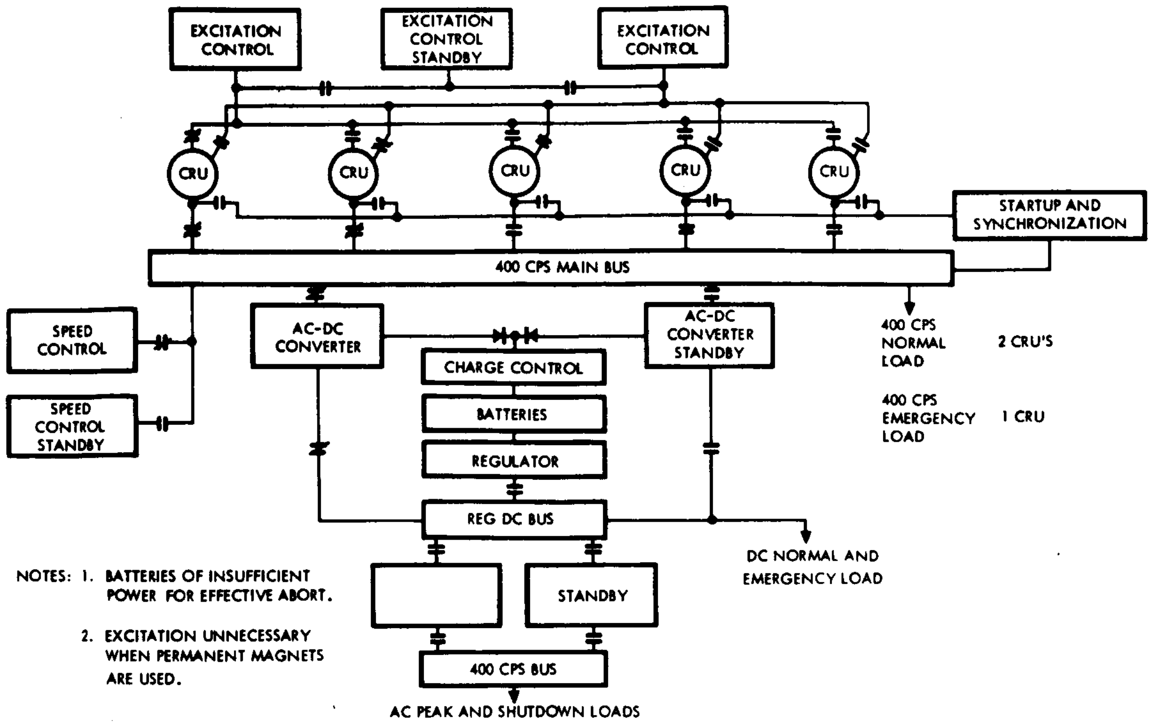


Figure 4.64. Electrical Power System, Block Diagram

Power levels of 4.2 kWe were established for backup requirements, considerable in excess of emergency levels of 2.8 kWe. Emergency power was dictated primarily by ECLS requirements and is defined as that power absolutely required for the safe return of the crew to earth.

Earth orbital power requirements are set at 3200 watts. With solar photovoltaic as the primary system, earth orbital needs presented additional problems. A specially strengthened array of 720 ft^2 had to be provided to withstand loads of 1 to 2 g's, randomly applied. This array can be included in the EPS as backup/emergency capability for use during the remainder of this mission.

A philosophy was established to provide a primary and backup/ emergency system. For the case of an isotope dynamic system the backup consists of cascaded thermoelectric compact converters which provide approximately 60 percent of the primary power required when used in place of the dynamic conversion equipment. The emergency system consists of a separate solar array sized for emergency requirements and deployed only in case of catastrophic failure of the total isotope primary system. This backup array would normally be stowed within the spacecraft and deployed only when required and the remainder of the mission would revert to the zero gravity mode to assure a high margin of safe return.

The preliminary work was accomplished at these levels with the isotope located in the EEM. Later study indicated that relocation of the isotope to the mission module was desirable to reduce spacecraft reentry weights. To simplify the power system concepts it was decided to size the isotope for the increased power demand of planet encounter. Consequently, a 7.0-kWe average power output is shown for the isotope system.

<u>The Rankine Conversion Cycle</u> — using Dowtherm, a floor wax derivative, is selected for the primary power source. A simplified schematic of the proposed power conversion system using an organic substance (Dowtherm A) as a working fluid is shown by Figure 4.65. This is basically a turbine prime mover operating in a closed Rankine cycle, with heat input to the cycle supplied directly to the working fluid in the boiler heat exchanger.

Heat regulation is accomplished directly in the condenser/radiator. The turbine wheel is mounted on an integral shaft with the alternator and the liquid pump. A regenerator heat exchanger is included to improve overall cycle efficiency, and an auxiliary radiator is used to control fluid temperature during the variations in heat output rate of the isotope. The peak temperature and pressure of the working fluid are 700 F and 107 psia. The minimum values are 275 F and 10 psia. The overall cycle efficiency is 12.2 percent, including all conversion losses and parasitic power consumption. Bearings are lubricated by the working fluid with the turbine-alternator pump assembly hermetically sealed.

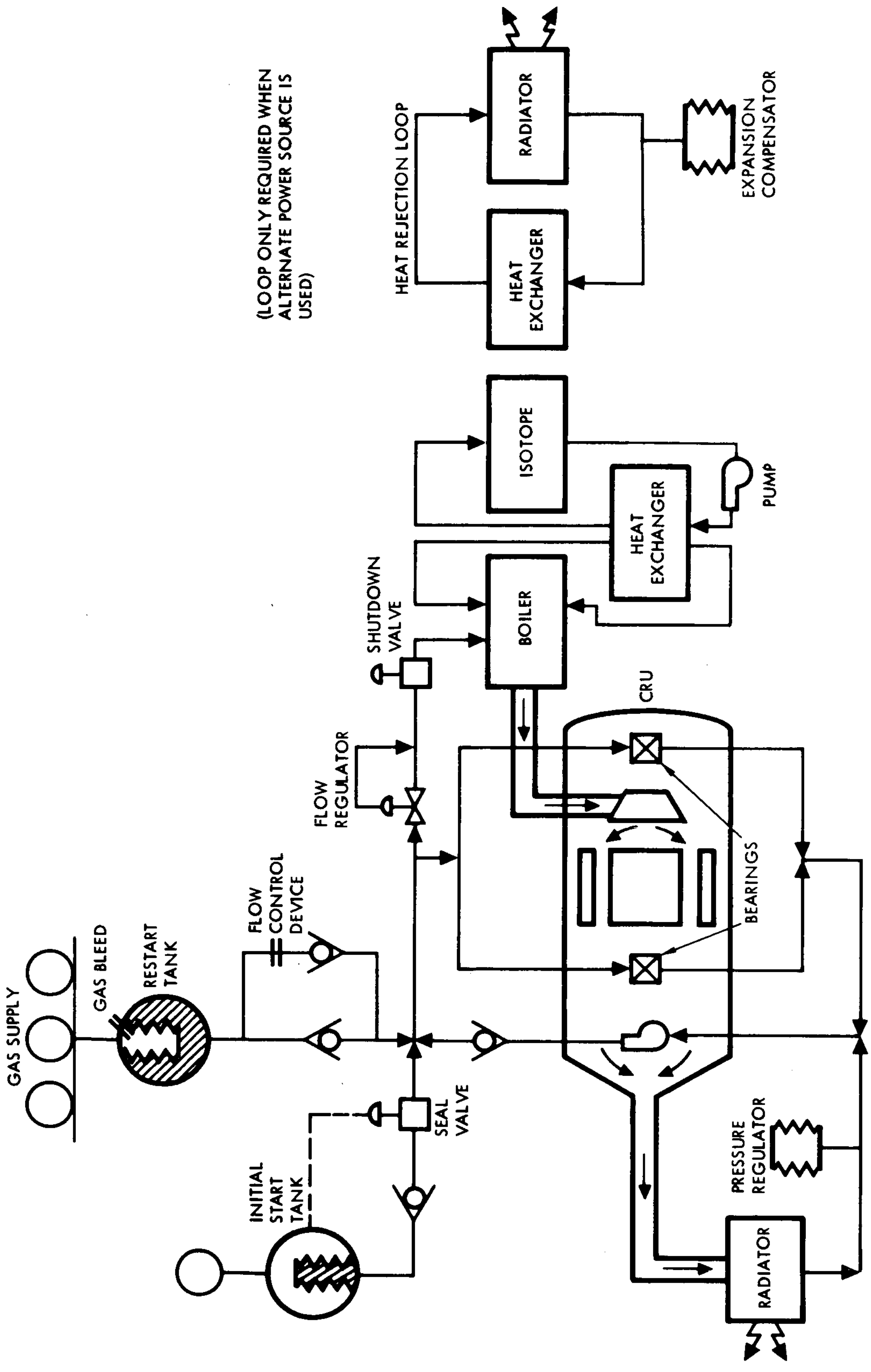


Figure 4.65. Radioisotope and CRU Rankine-Cycle System Schematic

Only Dowtherm A and mercury, which are comparable on a system weight basis, are competitive as working fluids for final selection. The SNAP 2 and SNAP 8 programs have provided much mercury performance data. Mercury requires a minimum radiator area, but its system temperature requirement is 1300 F, compared to 700 F for a Dowtherm-A system. From a reliability, safety and maintenance standpoint, lower temperature systems are more desirable and will result in minimum development risk and time. Also, the Dowtherm A radiator area requirements are within the 400 ft^2 available on the PM. For Dowtherm A as a working fluid, turbine design modification to the existing design is relatively simple since it is a single-stage design relying only on nozzle inlet design for change in power output. As a fluid, Dowtherm A suffers from gaseous decomposition at temperatures in excess of 700 F. However, with periodic maintenance this would not result in a serious problem. The isotope development for operating at 700 F is minimized. Existing isotope encapsulation techniques are proven adequate for this temperature.

The Cascaded Thermoionic System is proposed as a primary backup power source, the first of two backup systems to be used in the event of loss of the primary system, its schematic is presented in Figure 4.66. The compact thermoelectric converter consists of a closely packed array of SiGe or PbTe thermoelectric elements confined between a heat source and a heat sink. The heat source could be a hot NaK channel or direct radiation from isotope fuel capsules. A coolant loop is required to remove waste heat from the converter and transports it to a space radiator.

The compact converter themoelectric elements are also electrically connected in series and in parallel to provide the desired voltage and power. Interconnections between the series modules are used to provide the high reliability against open circuit failure.

An isotope source, fueled with Pu-238, is used to heat NaK in the primary loop. The NaK is circulated by an electromagnetic pump and maintains the hot junction temperature of the thermoelectric converter. Thermal energy passes through the thermoelectric elements in the compact converter to a second NaK loop that rejects waste heat by means of a space radiator. Isotope-thermoelectric systems may also be configured with direct radiating power converters. The weights and performance of compact thermoelectric converters are also representative for direct radiating power systems.

The inherent reliability, safety and simplicity of the TE system will permit a short and simple development cycle since all components are within the present state-of-art. The overall efficiency of this conversion will vary depending upon materials. Highest efficiencies can be relaized by cascading GeSi and PbTe so that both operate at their optimum temperatures. This

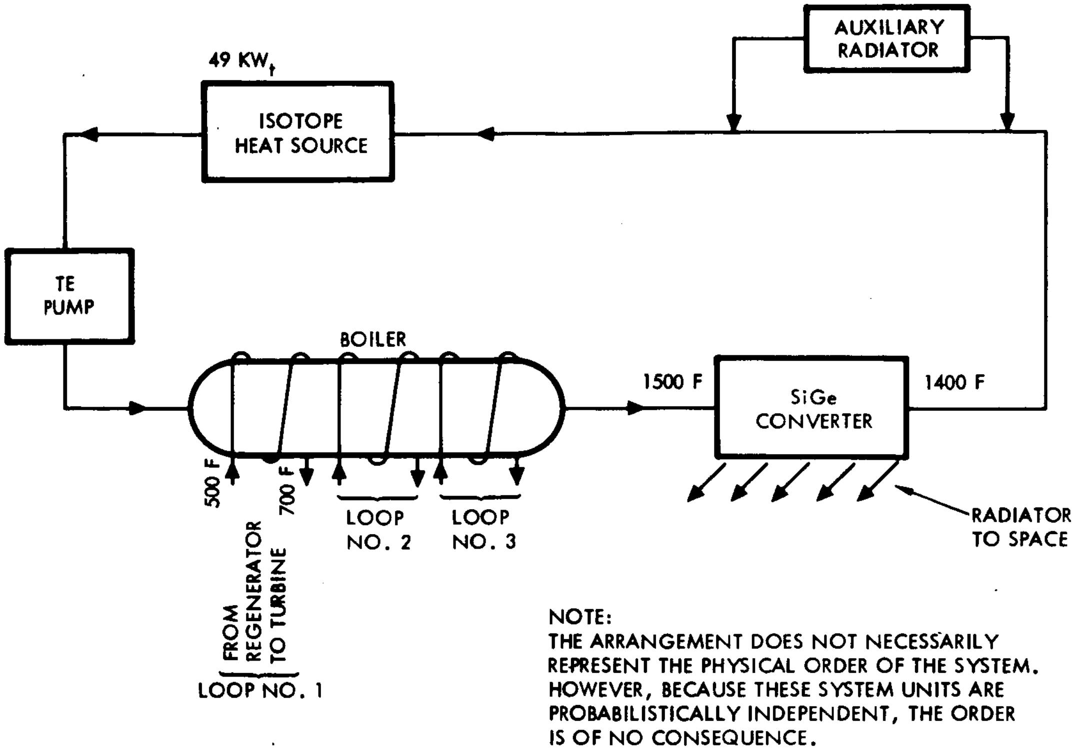


Figure 4.66. System Schematic, Isotope Thermionic Electrical Power Source

results in an overall gain in efficiency; i.e., from 7 to 8 percent overall efficiency compared to single element efficiency of 4 percent. At best, thermoelectrics will mean an increase in isotope requirement of about twice that for the dynamic systems if used alone. This becomes a critical consideration when isotope availability and cost is considered. These disadvantages of thermoelectric system preclude their selection as the primary power conversion system. However, since the radiator area requirements for the thermoelectric cascaded elements are competitive with the Rankine Dowtherm A and the thermoelectrical system was considered as a backup conversion to the dynamic CRU's, the combination offers a highly reliable power conversion design at a minimal weight penalty.

Reliability/Availability Analysis, Main Power Source

Since much of the components used in both the primary and secondary EP source are the same, the reliability analysis can evaluate them together. The component and subsystem generic and allocated failure rates are indicated in Table 4.42. These are considered in series for reliability logic and the basis for setting system availability requirements.

The combined rotating unit (CRU) is the component which will pace system reliability. A plot of system reliability as a function of CRU reliability and using two, three and four standby PCS loops has been made. The CRU failure rate was allowed to vary from 0.6×10^{-6} to 70×10^{-6}, while keeping other component failure rates constant at the indicated values; in addition, a very pessimistic meteoroid environment was assumed at a probability puncture of 0.972, Figure 4.67 summarizes this analysis.

If a failure rate of 2×10^{-6} per hour or less is expected from the CRU, a 1 active loop plus 2 standby loop system could be postulated for the 4 KW requirement. Replacement of the CRU is considered impractical at this time because of lack of good data, therefore, inactive redundance is suggested as a safe satisfactory alternative.

Normally two 3.5 kWe CRU's will make up the 7 kWe system.

The NaK from the isotope source heat exchanger will flow through the PCS boilers arranged in a series configuration. For greater than 4 kWe, two CRU's will be on the line, and two each of associated TE pumps and boilers will be active. This approach will provide the criticality I and II requirements at 0.999 for the power system. In addition, it can be seen that power system contribution to P_S at 3.5 kWe will be very high (about 0.999999) represented by one active plus four standby CRU loops in the recommended design. This is a worse case and the requirement could be as low as two standby loops, depending on the operational mode. Estimates are known to be pessimistic since usable data for the assessment was based on the mercury cycle tests a more deleterious working fluid.

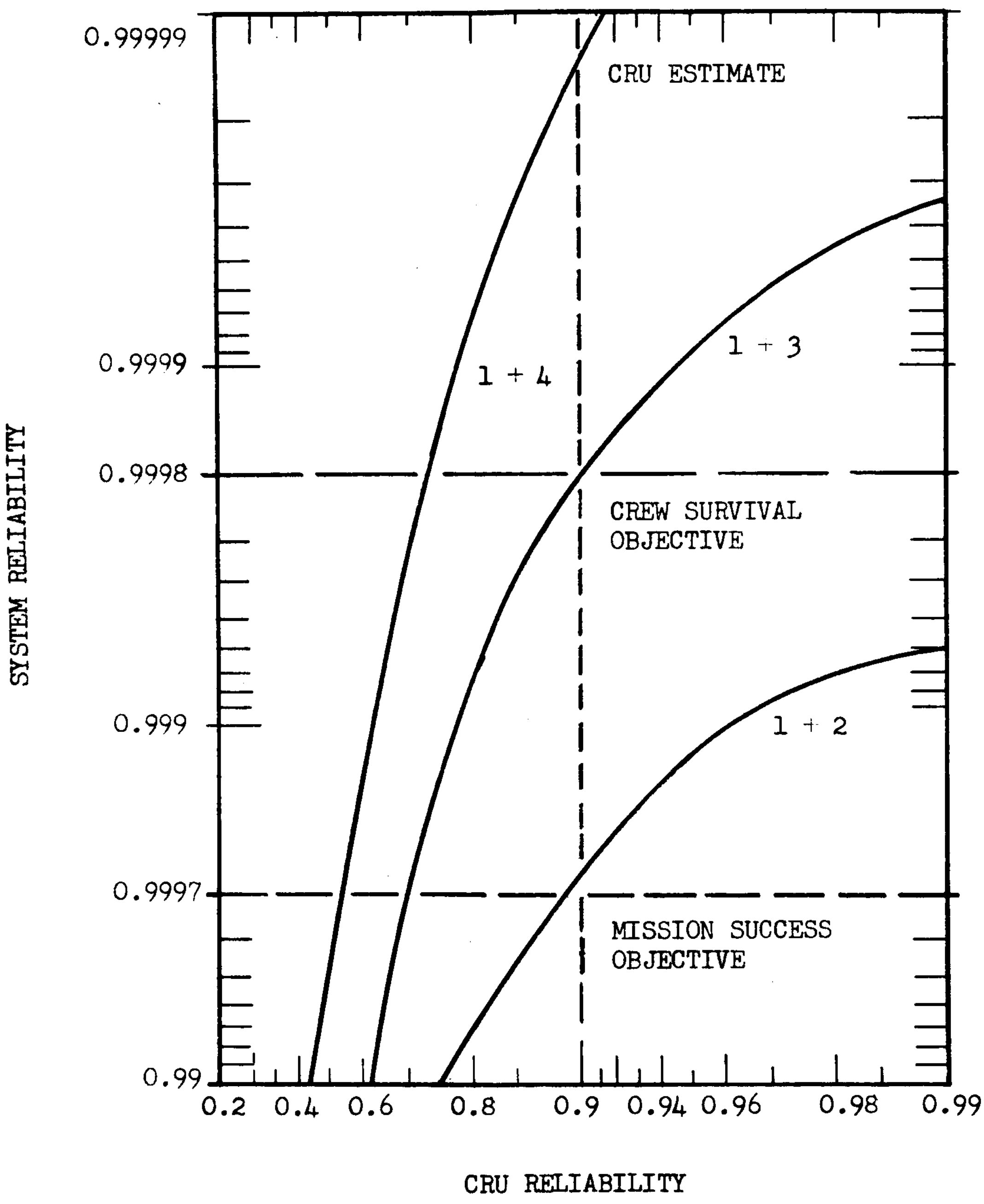


Figure 4.67. Effect of CRU Reliability on Total System Reliability

Table 4.42. Failure Rate and Reliability Estimate Rankine Power Source

Item No.	Component	State-of-the-Art General Failure Rates (x 10^{-6})		Maximum Expected Mission Failure Rate (x 10^{-6})			Reliability per Loop*
		Lower Limit	Mean	Upper Limit	Active	Standby	
1.	Alternator	0.033	0.7	2.94	0.05	-	
2.	Turbine	3.33	10.0	16.67	5.0	-	
3.	Pump (engine driven)	1.12	2.25	3.52	1.5	-	
4.	Bearings (3)	3 x (0.02)	0.5	5.5	0.086	-	
	CRU Total				6.636		0.902
5.	Radiator-condenser	2.21	5.0	12.5	3.418	0.618	0.618
6.	Boiler	1.0	4.4	19.6	1.45	0.2	0.2
7.	Inject tank	0.48	1.5	3.37	0.66	0.48	
8.	Regulator tank	0.89	4.25	15.98	1.25	-	
9.	Flow regulator	-	0.56	-	0.50	-	
10.	Check valve	0.112	2.3	4..7	1.0		
	PCS Loop Total				14.914	1.298	0.788**
11.	Radiator	2.21	5.0	12.5	5.0	1.0	
12.	Boiler	1.0	4.4	19.6	1.55	1.55	
13.	Expansion comp	0.89	4.25	15.98	3.6	1.0	
14.	Pump, EM	1.12	2.25	3.52	1.12	-	
	Heat Rejection Loop Total				11.32	3.55	0.840

*16,000 hr duty cycle, standby rate applies to inactive redundant loops only.

**Two loops provide a P_S of 0.955, three P_S = 0.982, four loops P_S - 0.997, five loops P_S = 0.9994 if all are active, otherwise $P_S >> P_S$ (active).

The other major weak link, the radiator, must be compensated for by oversizing it in the design stage to compensate for any tendency to clog or leak resulting from the meteoroid hazard.

Some maintenance may be possible on the Dowtherm Rankine system because of the lower operating temperatures and the working fluid (floor wax). Further, it may be desirable and indeed expedient to purge the system overboard and replace the fluid periodically to offset the effects of potential decomposure. In addition, it seems feasible to replace the CRU in a system of this nature, about 13 pounds are required for each cycle.

Reliability/Availability Analysis, Thermoelectric Backup Power Source

A study of a thermoelectric power source for a mission equivalent to the baseline mission used herein was conducted by AI/NAR and reported in Reference 4.18. The system design was estimated to achieve a reliability for a two-year mission when used as a primary power source of better than 0.98. This same system was upgraded to provide 5 kWe at a P_S of 0.99. The same design would meet the baseline mission emergency power level with a probability exceeding $P_S = 0.999$. The system requirements are given in Table 4.44. Maintenance is not expected to be required because of the low duty cycle and the high potential reliability in the emergency mode. Although the concept is obviously reliable and feasibility has been demonstrated, a test program is required.

4.9.2 Electrical Power Conditioning, Distribution and Control Subsystem (PCDS)

1. Functional Requirements

The electrical power distribution subsystem provides electrical energy, both unregulated and regulated, and electrical distribution consistent with the mission requirements.

All electrical loads applied by the EPS equipment are classified as essential and nonessential, this associated with crew survival and safe return of the spacecraft and those associated with mission success. Provisions have to be made for disconnecting the nonessential loads, as a group, under emergency conditions. Ordnance loads are normally supplied from a separate and independent battery source.

Distribution panels or boxes have to be enclosed to minimize hazards and to provide maximum mechanical protection. Switching and control may be accomplished by manually operated circuit breakers or contactors.

Table 4.43. Estimated Reliability for the Electric Power Source, Thermalelectric Backup System (16,000-Hour Duty Cycle Assumed)

System of Component	SiGe System	
	$\lambda \times 10^{-7}$	Reliability
<u>Heat Source</u>		
Isotope capsules	0.53	0.99907
Heat exchanger	1.2	0.9979
Shield	0.04	0.99993
<u>Primary Coolant System</u>		
Piping	0.4	0.9993
ECU		0.99996
Converter, electrical	1.0	0.9982
<u>Heat Rejection System</u>		
Piping	0.4	0.9993
ECU		0.99996
Radiator		
Performance	2.13	0.9963
<u>Meteoroid Requirements</u>		0.9957
Capsule safety ejection	0.95	0.9984
Across-the-line pump	2.4	0.9958
Performance reliability		0.984
Total Reliability	9.05	0.980

Table 4.44. Redundancy Recommendations, 5 kWe TE Power Source

Recommended Design Action	Two-Year Mission	Emergency Only
1. Redundant separate-source pumps in primary 2. Two active converter-heat rejection segments, plus one standby segment 3. Shutoff valves on heat rejection		
Performance reliability, TE system	0.9902	0.9999+
Required meteoroid non-puncture reliability per radiator segment	0.997	Depends on design
	0.99	0.999+

Power Return — The distribution system is normally a two-wire grounded system for DC and single-phase AC loads, and four-wire grounded system for three-phase AC loads; i.e., wire and buses should be employed as the return path for electrical currents rather than the spacecraft structure. The system negative and neutral should be grounded at one point only in the spacecraft and not be interrupted by any control or switching device.

The recommended DC and AC power characteristics for the baseline mission is as follows, as sensed at the load: (Reference 4.9.2-1).

a. DC Power

1. Steady-state voltage limits:	27.5 plus or minus 2.5 volts
2. Transient voltage limits:	21 volts to 32 volts with recovery to steady-state within one second
3. Ripple voltage:	1 volt peak-to-peak 30 to 15,000 cps

b. AC Power, if an AC power distribution system is used

1. Phases:	Three phases — displacement 120 degrees-tolerance, plus or minus two degrees
2. Nominal voltage limits:	
Steady-state	115 plus or minus 2 volts rms (average of three phases measured line to neutral)
Transient	115 plus 35, minus 65 volts rms recovering to 115, plus or minus 10 volts rms within 15 milliseconds and steady-state within 50 milliseconds.
Unbalance	2 volts rms (worst phase from average)
Modulation	0.5 percent maximum

3. Nominal frequency tolerance:

Normal	Negligible (synchronized to a master timing device)
Emergency	400 plus or minus 7 cps (loss of master timing)

4. Wave shape: sine wave

Maximum total distortion	5 percent
Highest harmonic	4 percent
Crest factor	1.414, plus or minus 10 percent
Maximum voltage	150 volts, 100 milliseconds

c. Unregulated 400-cycle AC power from individual inverters

1. Phases:	Three phases — displacement 120 degrees — tolerance, plus or minus two degrees.
2. Nominal voltage limits:	As inverted from the S/C 28 VDC distribution system without transformers.
Steady-state	21 plus or minus 2 volts rms (average of three phases measured line to neutral)
Transient	21 plus 5, minus 6 volts rms.
Unbalance	1 volt rms (worst phase from average)
Modulation	0.5 percent maximum

3. Nominal frequency tolerance:

Normal	Negligible (synchronized to a master timing device)

Emergency	400 plus or minus 7 cps (loss of master timing)
4. Wave shape:	Quasi square wave

Buses and electrical loads are to be selectively protected so that individual load faults will not cause an interruption of power on the bus to which the load is connected. A fault on a logic bus or an ordnance bus should not cause an interruption of power on the main buses. Similarly, a fault on a load circuit should not cause an interruption of the main buses, but should disconnect the fault at that circuit breaker closest to the fault.

The electrical equipment should be compatible, physically and functionally, with other spacecraft subsystems within which it is designed to operate.

The detailed design of a power conditioning, distribution, and control system can only be performed after the characteristics of the power source and the requirements of the spacecraft and mission are accurately known. For a solar cell power source, the power conditioning and distribution schematics are similar to those of an isotopic power system whose net product is regulated DC. The PCDS is basically the same with the exception of voltage regulators and battery charging circuits. If the net output of the isotopic power system is both DC and 400 cps AC, the inverters are, of course, eliminated.

DC-to-AC inversion is the major factor in overall life extension of Apollo type power systems. To alleviate this problem, three approaches were studied: (1) improvement in inverter reliability; (2) simplification of the inverter by changing the AC power characteristics to something other than an accurately controlled sine wave; and (3) elimination of AC power requirements. Westinghouse Electric (the manufacturer of the Apollo inverter), examined the first approach. The results indicate that significant improvements in reliability are feasible by (1) normal component improvement; (2) by upgraded components; (3) by circuit simplification; and (4) by incorporation of in-flight maintenance provisions. The other system reliability improvement approaches (mitigating or eliminating AC requirements) were considered by examining the characteristics of those components that are the primary users of spacecraft AC power.

The electric power generated can be DC and AC and can also be a function of power source selection, capable of variation within certain limits. Higher voltages are favored for reduced distribution losses, but are limited by source characteristics and available hardware. It is generally accepted to use 28 volt DC or 115 volt, three-phase 400-cycle AC in power systems. The 28 volt DC power distribution losses are relatively high and on several space systems 56 volt DC was adopted.

Electric power distribution systems are also limited by energy storage system selection. Usually electric storage batteries are used for this purpose. A matching of the main power source with the energy storage device and power distribution system is required. With a DC distribution system this is no difficulty. An AC generator output can be rectified with high reliability. On the other hand, an AC distribution system with static inverters in the storage battery output would limit the system overload capability and, due to the complicated inverter circuitry, impair system reliability. A DC electrical distribution system providing power within the spacecraft is preferred if the distribution line is not too long.

Based on SD studies in Reference 4. 19, it seems apparent that some equipment using AC power from separate inverters are preferred, eliminating the requirement for an AC distribution system, and simplifying both the distribution and reliability problems.

A DC power distribution system with localized identical inverters can be incorporated in the baseline spacecraft and reduce system weight. Spare inverters of a standardized plug-in type would increase system P_S to well above the ideal requirement.

2. PCDS Reliability Analysis

Since the EPS must operate full time, so must the PCDS function. This means a mission duty cycle of 16, 800 hours. However, this does not bear the same implication as an equivalent duty cycle on systems operating in a cyclic manner; continuous operation without transions is far less severe in terms of failure hazard.

The recommended power distribution concept, designed to assure a high degree of safety is presented in Figure 4. 68. This kind of an approach will minimize catastrophic loss of power. It also indicated that the requirement for highly conditioned power imposes a higher failure hazard. Therefore, during any design research, every effort should be made to relax the power conditioning requirements. Figure 4.69 presents the associated reliability logic diagram in simplified form as it was derived from the baseline mission. The logic block numbers correspond to the components given in Table 4. 45 as do the associated failure rate estimates which were taken from Apollo (Reference 4. 5).

The data indicates that two potential weak links exist in the function, the inverter, and the battery charger. These are responsible for the high hazard and low function reliability of 0. 787. This is obviously unsatisfactory and must be raised to above 0. 999.

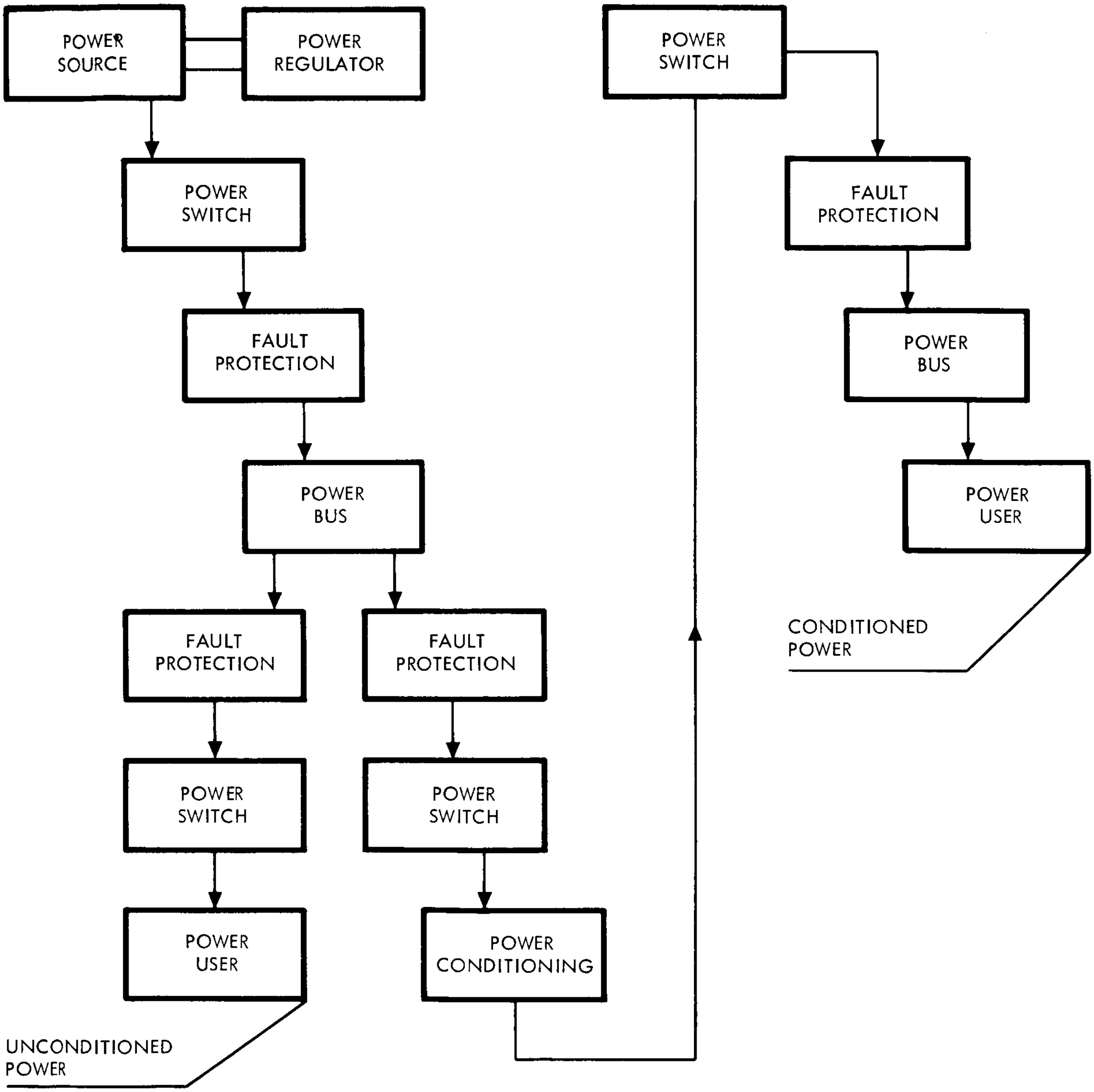


Figure 4.68. Typical Power Distribution System

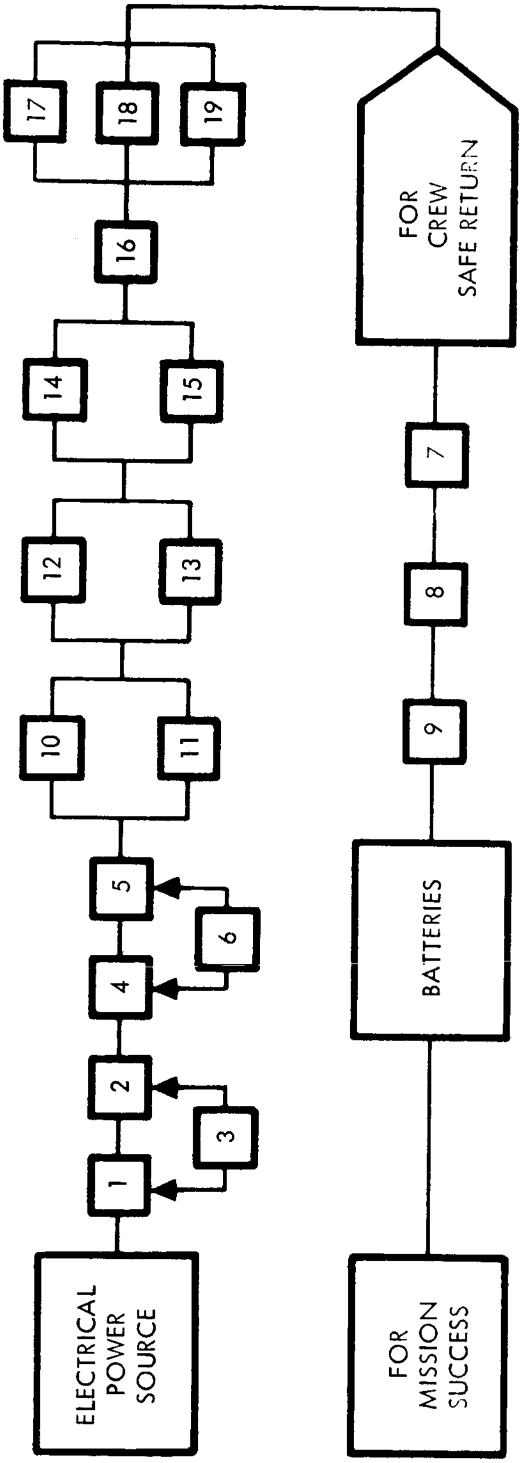


Figure 4.69. Electrical Power Distribution and Control Logic

Table 4.45. Electrical Power Conditions, Distribution & Control Subsystem, Reliability Estimates

Logic Block No.	Item	Failure Mode	Failure Rate (x 10^{-6} Hrs)	Duty Cycle (Hours)	Estimated Reliability
1	Entry battery A	No output	8.000	1.5 max	0.999964
	Shunt	Open	0.008	1.5 max	
	Circuit breaker	Open	0.120	1.5 max	
2	Entry battery B	No output	8.000	1.5 max	0.9999964
	Shunt	Open	0.000	1.5 max	
	Circuit breaker	Open	0.120	1.5 max	
3	Entry battery C	No output	8.000	1.5 max	0.9999964
	Shunt	Open	0.008	1.5 max	
	Circuit breaker	Open	0.120	1.5 max	
4	Battery bus A	Short-fail	0.005	1.5 max	≈ 1.0
5	Battery bus B	Short-fail	0.005	1.5 max	≈ 1.0
6	Battery bus C	Short-fail	0.005	1.5 max	≈ 1.0
7	Flight and post landing bus	Short-fail	0.005	16,000	0.999916
8	Battery relay bus	Short-fail	0.005		0.999916
9	Flight bus	Short-fail	0.005		0.999916
10	Main DC bus A	Short-fail	0.005		
11	Main DC bus B	Short-fail	0.005		
12	AC Bus No. 1	Short-fail	0.005		
13	AC Bus No. 2	Short-fail	0.005		
14	Pyrotechnic battery	No output	8.000	1.5	≈ 1.0
	Circuit breaker	Open	0.120	16,000	
15	Pyrotechnic battery	No output	8.000	1.5	≈ 1.0
16	Battery charger	No output	142.0	560	0.9921
	Shunt	Open-short	0.005	16,000	≈ 1.0
	Relay	Open	0.003	16,000	
	Switch, panel, rotary	Fail to operate	0.118/cs	16,000	
	Switch, panel	Fail to operate	0.003/cs	16,000	
	Circuit breaker	Open	0.120	16,000	
17	Inverter	No output	122.7	16,000	0.65*
	Switch, motor	Fail to operate	0.564	16,000	
	Switch, panel	Fail to operate	0.003/cs	16,000	
	Circuit breaker	Open	0.120	16,000	
18	Inverter	No output	122.7	16,000	0.65*
	Switch, motor	Fail to operate	0.564	16,000	
	Circuit breaker	Open	0.120	16,000	
19	Inverter	No output	122.7	16,000	0.65*
	Switch, motor	Fail to operate	0.564	16,000	
	Switch, panel	Fail to operate	0.003/cs	16,000	
	Circuit breaker	Open	0.120	16,000	
	System Total				0.787

*Any one of the three will permit the crew to return safely

3. Availability Analysis

An examination of the identified weak links demonstrate that the addition of one spare battery charger will raise the function P_S from 0.992 to 0.99997 and will eliminate this factor from further consideration as shown in Table 4.46 Having accomplished this, the data indicates that for all practical purposes the PCDS reliability is limited by the DC-AC inverter or an AC-DC converter reliability — even if all other components were perfect, no measurable improvement in reliability would be reliazed. It is clear from the analysis that the DC-AC inverter concept has to be improved for prolonged missions, or provisions for maintenance and repair must be provided.

The contemporary DC-AC inverter is a static inverter which provides a sinusoidal, 3-phase, 400-cycle output with high efficiency and low harmonic content. The reliability can be assessed for this unit based on failure rates and parts count of individual components, and verified by tests of completed assemblies. In order-of-magnitude the failure rates of the more ciritical components of this inverter are:

Filter capacitors	1.80×10^{-5} per hour
Power transistors	1.20×10^{-5} per hour
Power diodes	0.96×10^{-5} per hour
Micro-module preamps	0.75×10^{-5} per hour
Balance of components	7.55×10^{-5} per hour
Net Failure Rate	12.26×10^{-5} per hour.

From the net failure rate of the inverter we arrive at a MTBF (average life) of 8,156 hours. It is evident that improving the inverter reliability is not a simple matter of improving or avoiding the use of a few weak components; even if the four most critical component types were eliminated the MTBF would only be increased by 60 percent. Since the Apollo inverter is typical of contemporary hardware it would seem that repair is less practical than replacement of the unit, particularly if individual units are used. Table 4.50 indicates that six spare inverters are required to increase the function contribution to safe return to about 0.99964.

As a potential alternative, a square-wave interter can be built with a very small number of components; such an inverter will be more reliable, lighter in weight, and more efficient than a sine-wave interter. In view of the difficulties associated with providing sine wave AC power for prolonged missions, the necessity of a sine wave inverter must be reappraised and the rise of a square wave converter considered.

Table 4.46. Sparing Requirements Analysis, Electrical Power Distribution and Control System

Component	Expected Number Used	R_o	Spares Required	P_s	Spares Weight Unit in Pounds	Spares Weight Total
Inverter	3	0.65	6	0.99975	48	288
Battery charger	1	0.9921	1	0.999968	3.0	2.5
Subtotal (Crit II)		0.787	7	0.99964	-	290.5
Circuit Breakers						
for Safe Return	30	<0.1	16	>0.9999	2 to 5 oz	3
for Criticality II	20	<0.1	7	>0.9999	2 to 5 oz	1.4
Subtotal	50	<0.1	23	>0.99963	-	295

One serious objection can be raised on the use of square wave power — that is, the greater amount of electrical interference associated with it. Based on this analysis it is important to reassess the need for use of a centralized inverter system with AC power distribution. Separate square-wave inverters strategically located in the S/C could provide the necessary AC power with a decrease in electromagnetic interference and increased reliability. This approach was followed in the design of the Gemini S/C. There, however, every inverter was individually matched to the equipment it operated. This made servicing more difficult. A very good square wave inverter design is used for the Apollo S-II booster recirculation pumps. These inverters provide 42 volt three-phase AC from a 56 volt DC line without the use of transformers. They weigh less than one-third as much as the Apollo inverter for approximately the same output rating and have a failure rate 1.55×10^{-5} per hour as opposed to the 12×10^{5} for the centralized unit. Use of a small number of spare inverters would further improve system reliability and meet any reasonable goal. Thus, a survey of waveform requirements of subsystems using AC power shows that substantially all subsystems could operate on square-wave power.

The spacecraft components normally using or requiring AC electrical power are:

Lights. Fluorescent or incandescent. These operate as well or more efficiently on a square wave (of 400 cycles) than on a sine wave. Some of the lights in Apollo use DC and incandescent lights operate just as well from a regulated DC power source.

Electronic power supplies. These operate just as efficiently and reliably with less ripple in the output, using a square-wave source.

Induction motors. These may be made to operate on a square wave with slightly decreased efficiency; however, this is offset by the increased efficiency of the square wave inverter; 90 percent versus 80 percent for a sine wave inverter. The fact that an induction motor will be operated on square wave power should be included in its specification.

Instruments. Certain Instruments may require AC power of high sinusoidal purity. The amount of power required is not large, however, and may be avoided completely if such instruments are replaced by analog to digital converters and digital computation.

Electric Motors. The largest demand for AC power comes from electric motors. Three-phase induction motors are used for their self-starting capability and, because they require no brushes or switches, they also operate reliably at good efficiency. Reliability of the three-phase inverter-motor combination should be improved for long missions. All electric motors operate with alternating current. Those motors which invert DC to AC power are called DC motors. The commutator is nothing more than a mechanical state inverter built into the motor. Such motors are built by several companies for space programs. Integrating the inverter with an AC motor lends itself to arrive at special motor characteristics at the cost of weight volume and efficiency. Overall system reliability would be reduced by the use of so-called brushless DC motors instead of separate inverters with separate AC motors.

Battery Charger Circuits. The state of charge of electric storage batteries is controlled by battery charging circuits. The unmanned S/C orbited today which utilizes a solar-cell power source, employs batteries and battery chargers. These battery chargers use excess power from the main power source to bring the batteries to the desired state of charge and regulate the rate of battery charging. The battery chargers, in general, use a simplified inverter circuit combined with voltage regulators and sometimes, current regulators. Reliability of basic battery charger circuits is high.

Communication systems are operated from the DC power line in unmanned spacecrafts. The selection of 400-cycle AC sine-wave power in the Apollo S/C was based on the requirement of synchronizing the different parts of the data system with the AC line. If DC power is used, a new synchronizing function will have to be provided. This is solved for the lunar excursion module and for unmanned S/C by providing a centralized timing pulse for these functions, as is recommended for the baseline mission.

Navigation and Control Systems. Several navigation systems are suitable for space application requiring only DC power. The Autonetics N-16 system and the AC Sparkplug Carousel Mark V systems were developed for manned aircraft and have reliability goals compatible with long-duration space mission requirements.

Other spacecraft subsystem and components normally call for AC power, but, in each case, DC can be used directly, or through a small local inverter.

Other parts which enter into the P_S estimate are included in the respective subsystem or the power source with the possible exception of the circuit breakers. As indicated in Table 4.46, there are about 30 which affect the Criticality I function (P_S), and 20 more which affect Criticality II — the less essential function. Using the normal failure rate estimates which are based on total on time rather than use cycles produces a very pessimistic estimate of the supporting spares required to raise the P_S contribution to over 0.999. The results are listed in the referenced table along with the respective weights. The replaceability of these components are demonstrated by a typical installation depicted in Figures 4.70 and 4.71.

4. PCDS Conclusions

The contemporary concepts for the PCDS will permit a safe and successful mission where the recommended maintenance provisions are included. The number of maintenance actions (spares required) will probably not exceed 30, and the spares weight will probably not exceed 295 pounds in achieving a contribution to the P_S of better than 0.9996.

4.9.3 Batteries

Note: Much of the information used in this section was furnished by Eagle Pitcher Company through Reference 4.20.

1. Battery Applications

The selected baseline mission as well as any other manned planetary mission is expected to carry batteries to augment the main power source and for special applications. Contemporary batteries of many types have been used successfully for periods far in excess of the baseline missions; for example, the three-year auto battery. Therefore, it seems evident that contemporary hardware such as that used for Apollo will meet the needs of the near future planetary missions. To substantiate this premise, the analysis herein was conducted.

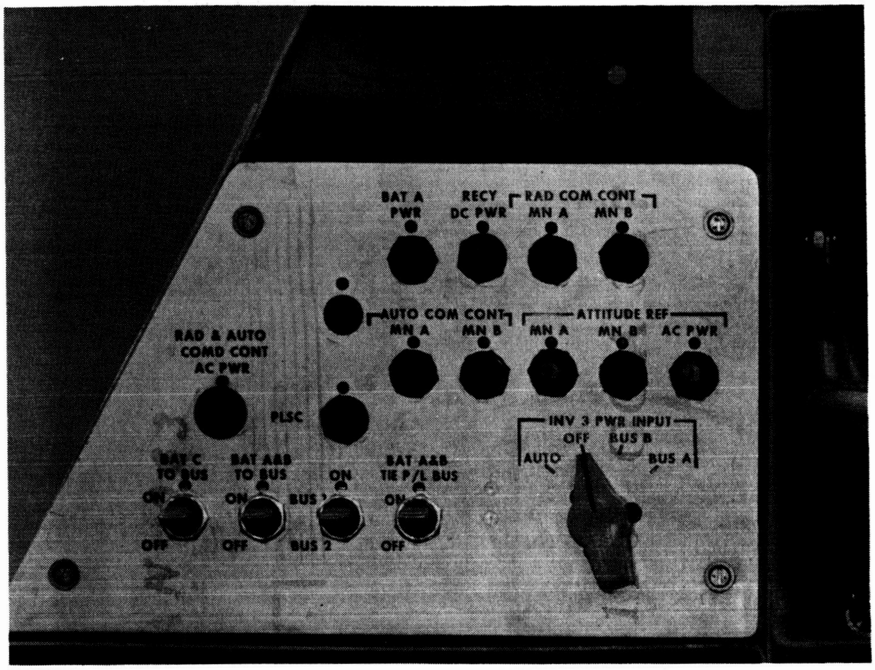


Figure 4.70. Power Programmer Control Panel in Place

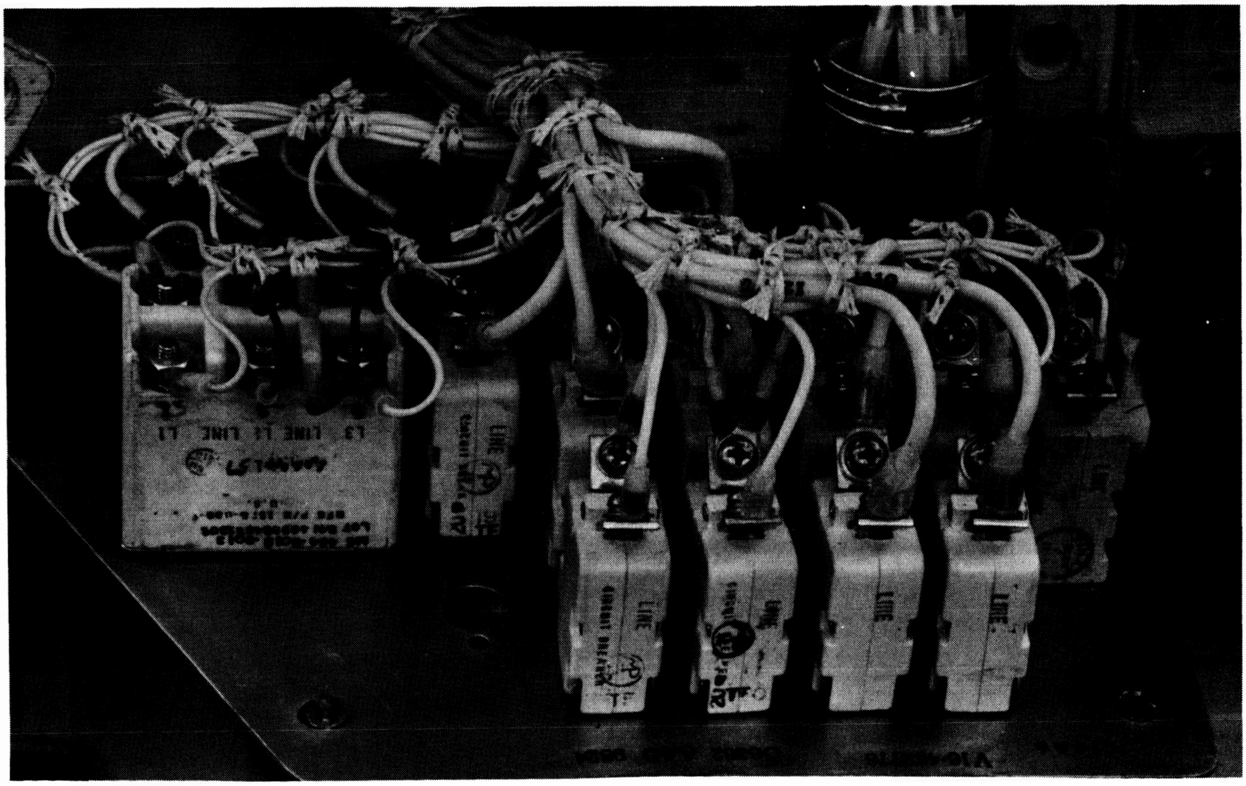


Figure 4.71. Power Programmer Control Panel Removed for Maintenance

For the baseline or any extended Mission of approximately a 700-day duration the life of the batteries should be in excess of two years. A considerable effort has been devoted toward examination of existing hardware from the APOLLO Program as well as existing hardware of other space programs and the possibility of utilizing new hardware of various electrochemical systems.

There are at least two distinct power requirements for the baseline mission, the peaking battery and the reentry battery. It appears to be the desirable to separate these into two distinct battery packages because of their diverse requirements. For the peaking battery, total ampere-hour requirements are of lesser importance than voltage regulation considerations. Silver-zinc, silver-cadmium, and nickel-cadmium batteries were examined and, in view of the intended mode of operation and the reliability require, it is recommended that a nickel-cadmium battery be used for this application.

For the reentry battery and some pyrotechnic batteries, the mode of operation would be such that virtually two years in the inactive state would proceed actual battery utilization. It appears that to achieve maximum reliability, this inactive period should be accomplished with the battery unactivated, that is in a dry-charged state. There are two methods of achieving this; one is the conventional remotely activated battery that is described in Reference 4.20A; the other is a dry-charged manually activated battery that would be activated by the spacecraft personnel prior to reentry. For this specific requirement, a design concept has been evolved and demonstrated which would permit such activation to be performed with a minimum of physical handling problems, since there may be some limitations in the ability of the personnel to perform intricate tasks in the projected environment. The exact physical configuration would be determined by the installation position of the battery, the resultant accessibility, and the degree of mobility possible by astronauts. The concept is discussed in detail in a subsequent section.

2. Reliability Assessment

Table 4.47 presents the reliability estimates for various battery types under the baseline mission conditions and as a function of wet stand/use life. From these data, it is evident that selection of the battery type, application, and operational concept must be matched. But, in so doing a reliable combination can be found.

Table 4.47. Reliability Estimates for Various Battery Types

Time Months	MAR	Ag-Cd	NiCd	EAP	MAP
1	0.999	0.9998	0.9999	0.998	0.999
6	0.980	0.998	0.9905	0.997	0.999
12	0.800	0.980	0.999	0.996	0.999
18	0.500	0.900	0.998	0.995	0.998
24	-	0.800	0.997	0.994	0.997
ESTIMATED WATT-HOURS PER POUND	60	36	18	24	72
RELATIVE WEIGHT	120%	200%	400%	300%	100%

NOTES: MAR type is a manually activated rechargeable silver-zinc battery
GAP type is a remotely activated primary silver-zinc battery
MAP type is a manually activated primary silver-zinc battery

The reentry batteries affect crew safe return (Criticality I) since they provide the EEM with its only source of power, after separation and during the reentry and recovery phases. However, from Table 4.47, it is evident that the reliability of NiCd battery will exceed 0.9999 for the short 1- to 2-hour reentry period.

The peaking batteries contribute to crew safe return only in that they provide temporary power during main power outages for repairs. Since this is the case, the estimated 0.997 mission reliability could fulfill that requirement without change. However, the mission success contribution, a lower criticality, will only be about 0.994 which may be considered too low.

3. Availability Analysis

To assure reasonable probability of mission success (Criticality II) both proper situations of the battery type and operational concept is required. To do this, each application must be considered separately.

A. Peaking Batteries

From an analysis of Table 4.47 relative to the requirements of the peaking battery, it become obvious that only the nickel-cadmium battery has acceptable reliability after two years of operation. Conversely, there is a very small chance that the silver-zinc rechargeable battery (if in a charged state) would operate normally after two years. The silver-cadmium battery, although having a greater probability of success, is also less desirable than the nickel-cadmium. The remotely activated battery would not meet the functional requirements of a peaking battery. The nickel-cadmium battery appears to achieve the maximum reliability, and with redundant units, will meet the projected requirement to exceed a P_s of greater than 0.999 for the function. Some design effort will be required to integrate the battery requirements with the power system and charge controls. Because the considerable experience with batteries and with automatic charge control equipment accumulated on the GEMINI, APOLLO, and other space programs, this task has been accomplished and an acceptable reliability achieved utilizing the nickel-cadmium system. Since voltage regulation is the principal concern, the watt-hour per pound estimate for a peaking battery is misleading, and in this application is a secondary consideration.

To achieve two years of life for the silver-zinc rechargeable peaking battery concept, each battery would require a great amount of separation and would be significantly derated until any weight advantage it might have originally presented for short missions will be considerably reduced. As might be expected, the silver-cadmium system is somewhat better in energy density, but still falls short of the nickel-cadmium battery in reliability. Since voltage regulation (internal impedance) is a primary consideration here, the nickel-cadmium battery is the best suited, not only for its excellent voltage regulation capabilities, but also for its proven operational reliability throughout periods measured in years. The failure modes of the nickel-cadmium battery are internal cell shorting, loss of electrolyte, memory effect, and externally induced physical damage. Internal shorting can be almost eliminated as a probable failure mode if proper manufacturing and inspection procedures are adopted. Eagle Pitcher has nickel-cadmium batteries in space on classified programs that have been operating successfully for years. The loss of electrolyte can be prevented by using hermetically sealed cells and paying particular attention to charge control. Electrolyte is frequently lost because of overcharging. Sealed cells can tolerate a nominal continuous overcharge rate without damage. Since most sealed cells use steel cases, the construction is quite rugged, and the possibility of damage is limited to a possible crack in the ceramic terminal seals.

Two peaking batteries are required for the normal peaking load, one battery will provide any emergency requirements necessary to assure crew safety during any main power outages. Therefore, with one spare the crew safety contribution (P_s) is well above 0.9999999 and there would be less than one chance in 10^6 of there being no batteries at any time during the mission. This adds less than 100 pounds to the spares complement required.

B. Reentry Batteries

For the reentry battery the problem is considerably different. Again referring to Table 4.48, the reentry battery requirements are such that acceptable reliability is possible with two forms of the nickel-cadmium battery, either the remotely activated primary or the manually activated primary battery. For this application, the watt-hours per pound criteria is a valid selection criteria and the primary manually activated and the remotely activated primary silver-zinc batteries both present significant advantages over the nickel-cadmium battery.

These batteries are squib activated and when stored dry and at nominal temperatures, exhibit a high degree of reliability. And the remote activation concept for these batteries are inherrently less reliable than their manually activated counterparts because of the additional failure modes associated with the activation device. However the reliable will be acceptably high.

Manually activated primary silver-zinc battery normally has a wet life measured in days as compared to the 1-hour to 5-hour-life requirement. However, they may be designed in a manner to extend this period to two weeks. By activating these batteries manually by adding electrolyte a few hours or several days prior to the intended usage, the highest degree of reliability and the lowest weight and volume could be realized, and the mission objectives achieved without any redundancy or repair. The problems of requiring an astronaut to perform the task just prior to reentry can be minimized through application of the technique presented in Figure 4.72, a simple method of activation in space.

The cell case would have a filling port that would be threaded on the inside to mate with the outside threads of a cylindrical polyethelene electrolyte reservoir. The reservoirs would be screwed in against a stop which could be overcome by twisting the reservoir with greater force than originally required. At the bottom of the chimney would be a sharp blade-like device which would cut the seal over the mouth of the threaded neck of the electrolyte reservoir. This would permit electrolyte to run into the cells. The reservoir, being somewhat flexible, could be "pumped" so that activation in a vacuum and/or zero-g condition would be possible. Capillary action would hold the electrolyte into the separators.

NASA experience in operating batteries for extended periods under zero-g conditions present ample evidence that no problems will be encountered with containing the electrolyte within the cells. Because batteries will gas upon activation and during operation, a threaded, molded boss would be incorporated in the end of the cylindrical electrolyte reservoir. After activation, a valve with a sharp blade-like device on the bottom would

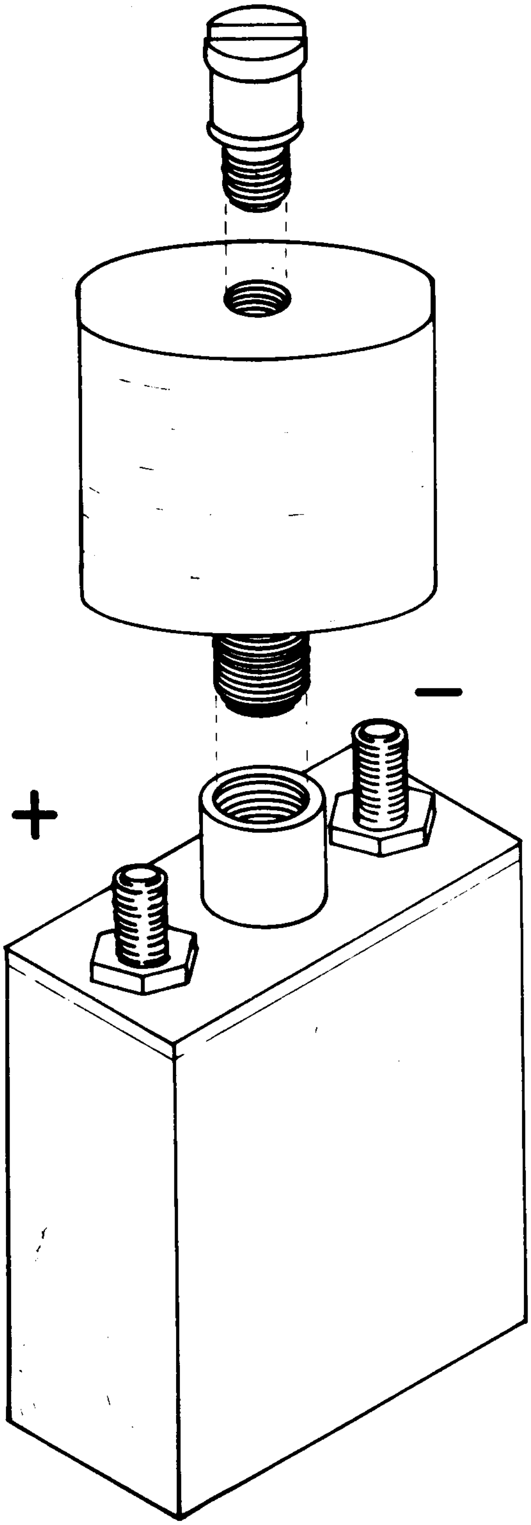


Figure 4.72. A Manually Activated Battery for Space Applications

be screwed into the molded boss. The sharp blade would puncture the electrolyte reservoir and the cell would then be able to vent through the reservoir and the valve. There may be other possible locations for the valve, should there be physical reasons that would preclude such a configuration.

4. Conclusions and Recommendations on Batteries

This analysis demonstrates what inherently seems to be true, i. e., contemporary batteries will meet all the extended mission requirement and with minor packaging changes to meet the specific mission requirements.

The nickel-cadmium cell will meet the requirement without change and only one spare battery for mission success assurance. This amounts to an additional 100 pounds of weight.

The manually or automatically activated silver zinc cells will meet both the reentry battery and pyro battery requirements. Minimum modifications to contemporary designs have already been tested and proved.

4.9.4 Conclusions and Recommendations on Electrical Power

The analysis the electrical power system requirements indicate that a system whose logic is as reflected in Figure 4. 73 will meet the baseline mission objectives and surpass them. The probability of no failure which cannot be repaired will exceed 0. 9995. The supporting spares required will involve:

1. For Criticality I: 16 spares amounting to 3 pounds plus the recommended redundancy within the power source, i. e., 2 redundant CRU loops, a cascade TE system, and some solar voltaic cells.

2. For Criticality II: 15 additional spares amounting to about 392 pounds, most of which is associated with the batteries and power conversions.

The results of these provisions assure a probability of crew safe return contributions of over 0. 999999, or there is less than one chance in 10^6 of a power system failure causing the loss of the crew.

Work is required in the area of isotope power sources to demonstrate their performance over long periods of time; and to develop an economical process for extracting the recommended isotope (Plutonium 238). For details on test and study requirements, see the individual functions.

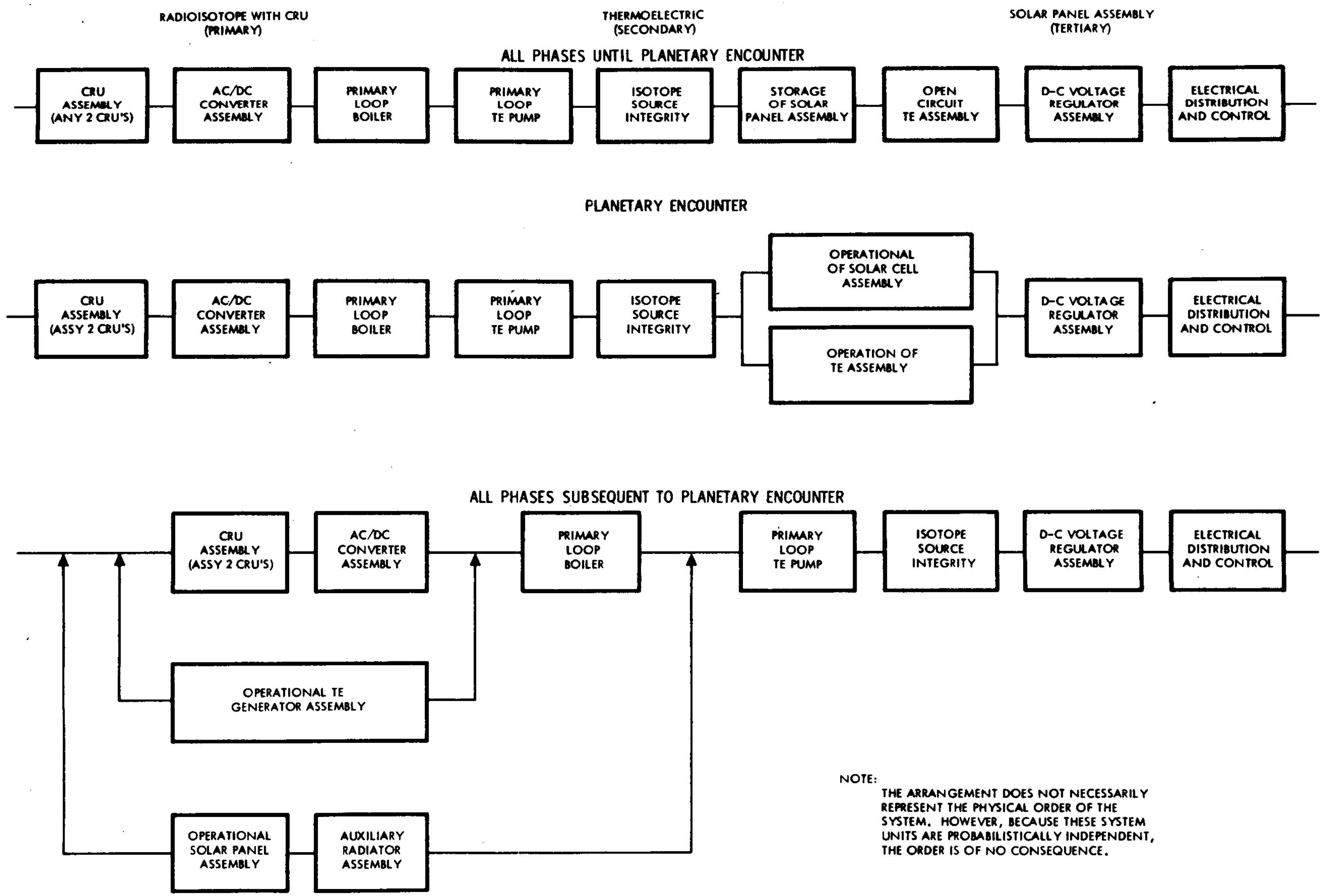


Figure 4.73. Radioisotope CRU With Either Solar Panel or Thermoelectric Generator Peak or Emergency Backup, Logic Diagram

4.7 THE CENTRAL TIMING SYSTEM

Note: Much of the data used in this section was provided by the General Time Corp., through Reference 4.21.

4.7.1 Functional Requirements

The functions of the Central Timing Equipment (CTE) are straightforward since it provides the master time reference for all mission systems. The outputs appear in two forms, Time Accumulation and Frequency Division. The CTE is a single component of about a ten-inch cube; internally, it has four module boards.

The function is required to facilitate navigation measurements and apply a thrust vector. It may be updated via the Up Data Link system. Complete loss of the function would impair crew ability to return safely, it is considered a Criticality I function.

The CTE consists of four functional subsystems:

1. Power supply
2. Oscillator
3. Frequency divider
4. Time accumulator

The CTE power supply subsystem consists of two separate identical pulsewidth modulated power supplies, each capable of handling full load. The power supplies are fed by nominal 28 VDC lines. See the functional block diagram of Figure 4.74.

The oscillator is a highly stable device and can operate in a primary or secondary mode. The primary mode is an internal 1024 kHz oscillator. In the secondary mode the CTE is driven by an external 1024 kHz signal. Both signals are double, shaped, and fed to a divide-by-four majority logic network. The oscillator output is 512 kHz.

The frequency divider is composed of successive two-out-of-three majority logic networks which divide the 512 kHz signal down to one Hz.

The time accumulator subsystem is composed of two sets of outputs. The first of these is a serial time code output conforming to the IRIG B format. The second is a set of parallel time code outputs.

Physically, the CTE package configuration weighs approximately 10 pounds and has a volume of approximately 225 cubic inches.

The present Apollo CTE is completely qualified for the baseline mission except to assess its life expectancy. Tests to date reveal no potential weakness.

4.7.2 Reliability Assessment, CTE

The reliability goals for the Apollo CTE have been established for a period 14 consecutive days under specific environmental conditions which are commensurate with the baseline mission requirements. These are as follows:

> For any set of Time Accumulator outputs, the probability of success shall be 0.999.
>
> For any single Frequency Divider output, the probability of success shall be 0.999999.

The CTE utilizes various types of internal redundancy to achieve the reliability goals. The power supply is a one out of two parallel arrangement whereby each power module is capable of handling the full load in the event of failure of the other.

The oscillator section utilizes dissimilar parallel redundancy. This arrangement allows the internal crystal oscillator to phaselock with an externally supplied signal of the same frequency. In the event of failure of the external signal, the internal oscillator continues to provide the correct signal to the frequency dividers without any loss of count.

The frequency divider section uses two-out-of-three majority vote logic. The output from the oscillator or previous divider is fed to three independent divider chains. Any two of the three chains must have identical frequencies and be in phase for a proper output for that divider stage to be present.

Additional component part redundancy is used in various circuits to guard against shorts, or opens, depending upon the criticality of the part to successful circuit operation.

The subfunctional level reliability logic for the CTE is comparatively simple as expressed by the logic of Figure 4.75. The reliability of the total function for the baseline mission is at least 0.978; that of the subfunctions are as indicated.

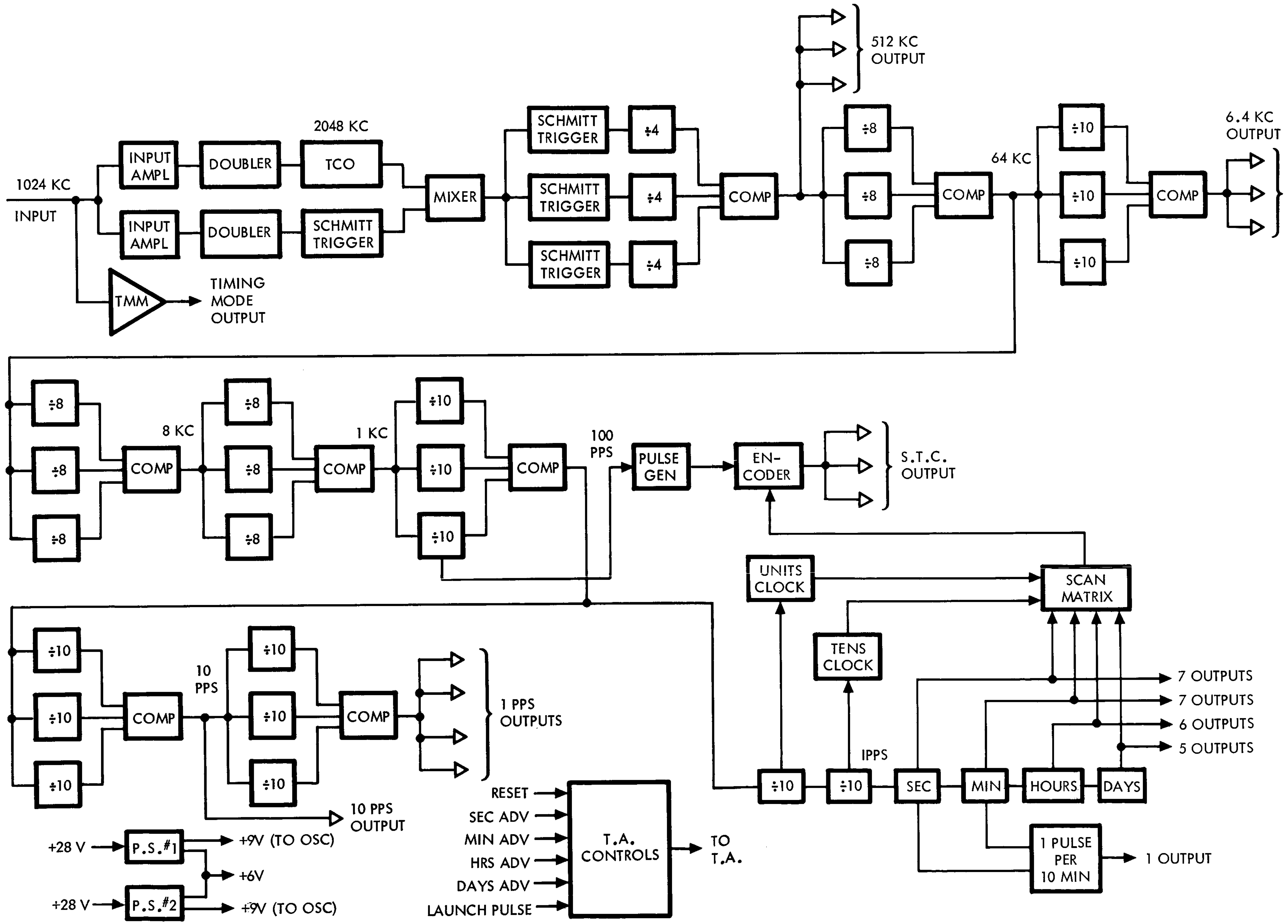


Figure 4.74. Functional Block Diagram Central Timing Equipment

FOLDOUT FRAME 1

FOLDOUT FRAME 2

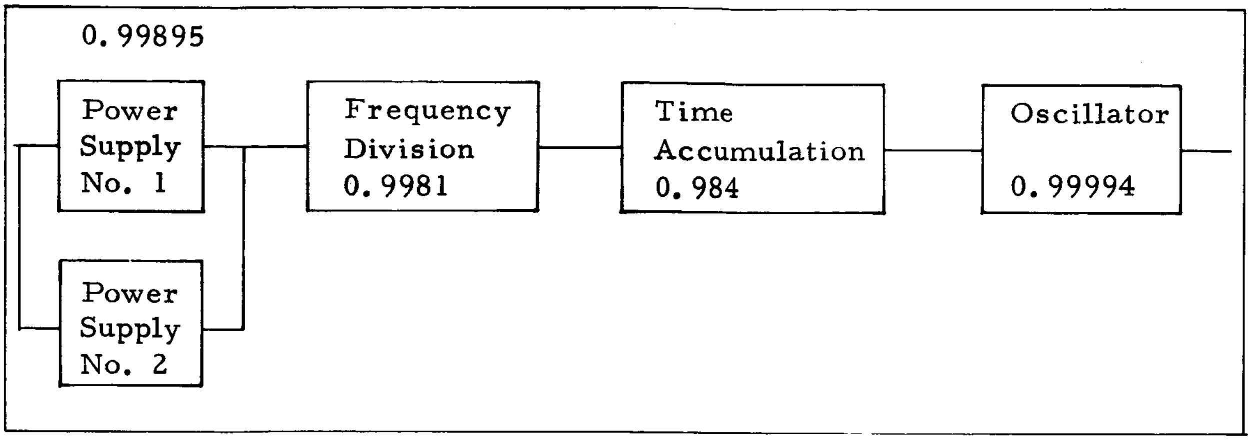


Figure 4.75. Reliability Logic, CTE, Subfunctional Level

The reliability is obviously too low in spite of the internal redundancy.

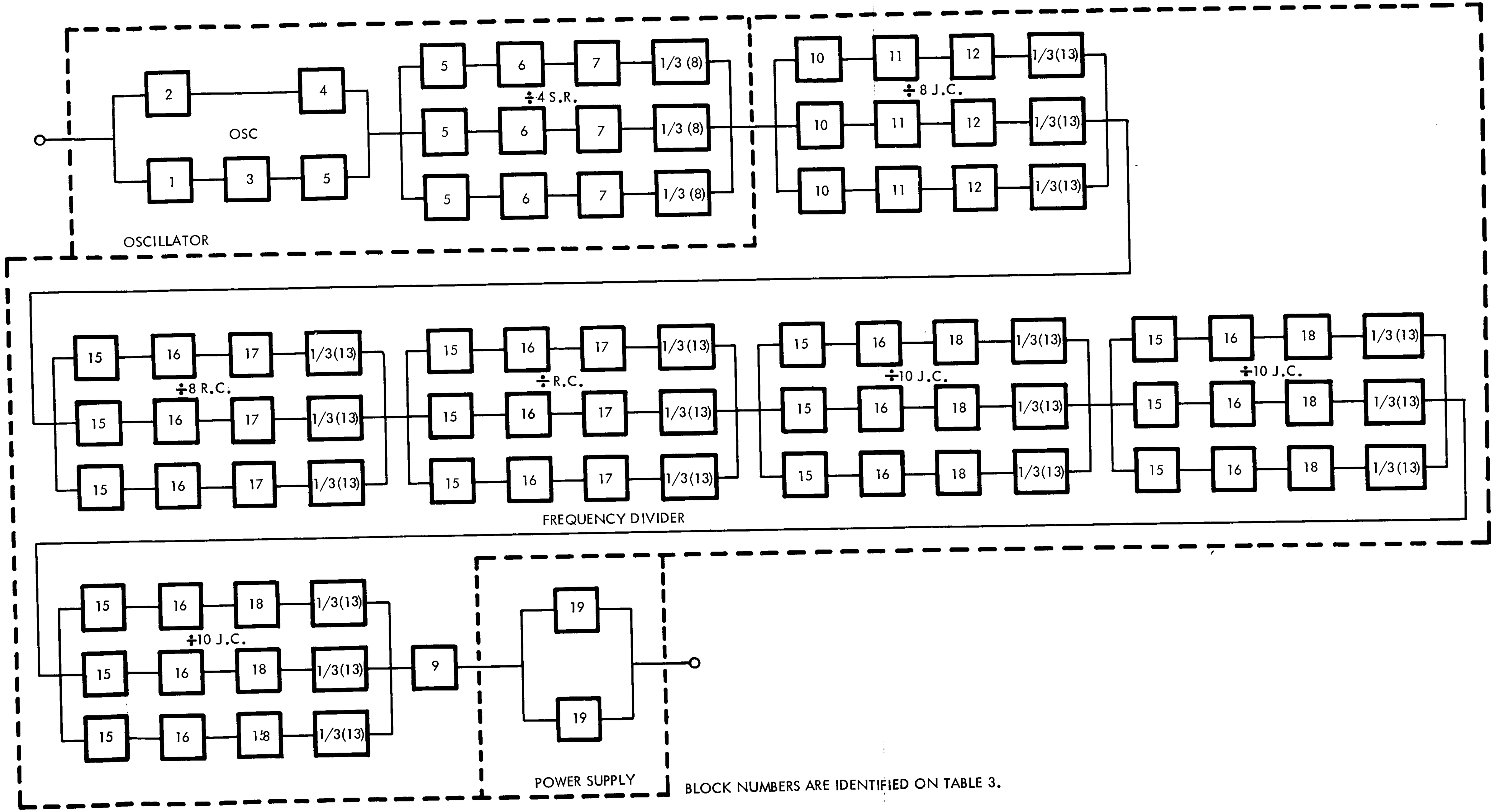
Figure 4.76 shows the reliability block diagram for any frequency divider output. The on-Hz output is the worst case frequency divider output since it represents the output preceded by the greatest number of dividers. Its reliability for the baseline mission is only 0.997. All other frequency divider outputs have a correspondingly higher reliability.

Table 4.48 Circuit List

Reference No.	Circuit	Circuit Failure Rate (λ)
1	Input amplifier	0.117×10^{-6}
2	Doubler "A"	0.080
3	Doubler "B"	0.082
4	Temperature compensated oscillator (TCO)	0.639
5	Schmitt Triggers A, B, C, D	0.073
6	Reset "R1"	0.041
7	÷ 4 shift register	0.230
8	Comparator "C1"	0.222
9	Buffer "B1" (Quad)	0.122

Table 4.48 (Cont)

Reference No.	Circuit	Circuit Failure Rate (λ)
10	Reset "R2"	0.028
11	Counter Driver "D1"	0.043
12	÷ 8 Johnson counter	0.050
13	Comparator "C2"	0.222
14	Comparator "C3"	0.222
15	Reset "R3"	0.028
16	Counter driver "D2"	0.041
17	÷ 8 ripple counter	0.030
18	÷ 10 Johnson counter	0.060
19	Power supply	1.930
20	Seconds/minutes counter	0.080
21	Seconds x 10/minutes x 10 Counter	0.054
22	Hours counter	0.128
23	Days counter	0.092
24	Time accumulator buffer	0.010
25	Time accumulator ÷ 100	0.120
26	Time accumulator reset and update	0.152



BLOCK NUMBERS ARE IDENTIFIED ON TABLE 3.

Figure 4.76. Frequency Divider Output, Reliability Block Diagram

FOLDOUT FRAME

FOLDOUT FRAME 1

FOLDOUT FRAME 2

Table 4.49. CTE Frequency Divider Subfunction Reliability Estimates

	Duty Cycle	Reliability (Individual Networks)
512 kHz divider network	16,800	0.99852
64 kHz divider network	16,800	0.999968
8 kHz divider network	16,800	0.999975
1 kHz divider network	16,800	0.999975
100 Hz divider network	16,800	0.999965
10 Hz divider network	16,800	0.999965
1 Hz divider network	16,800	0.999965
Quad buffer network	16,800	0.9999996
Oscillator	16,800	0.99994

To calculate system reliability, all of the frequency divider networks are considered in series. For the worst case output (R_{1Hz}):

$$R = Rosc \cdot R512 \cdot R64 \cdot R8 \cdot R1 \cdot R100 \cdot R10 \cdot R1 \cdot Rquad \cdot Rps$$

$$= (0.99994479)\,(0.998521)\,(0.9999678)\,(0.9999747)\,(0.9999747)$$

$$(0.9999651)\,(0.9999651)\,(0.9999651)\,(0.99999962)\,(0.998949)$$

$$R = 0.99723$$

For any set of time accumulator outputs, the reliability is given by:

$$R = Rosc \cdot R512 \cdot R64 \cdot R8 \cdot R1 \cdot R100 \cdot Rta \cdot Rps$$

$$= (0.99994479)\,(0.998521)\,(0.9999678)\,(0.9999747)\,(0.9999747)$$

$$(0.9999651)\,(0.986224)\,(0.998949)$$

$$R = 0.98356$$

4.7.3 Availability Analysis, CTE

The reliability analysis reveals that the present system configuration does not meet the requirements for a 700-day mission.

One approach to this problem would be to have one or moer spare CTE's on hand in the event of circuit failure. This approach requires consideration of the following:

1. If it were desired to make the two CTE's functionally redundant, both operating simultaneous, methods of load sharing need be devised and some redesign would be required to accomplish this objective.

2. If it were desired to provide standby redundancy with a second CTE, a method of switching is required. This switching system may be complex due to the number of output lines required and this may introduce some unreliability.

3. If the second CTE were treated as a shelf-spare, timing functions would be interrupted in the event the spare was interchanged with the original CTE. This can be handled through use of the Up-Data Link.

Replacing the CTE, when failed, will raise the contribution to P_s to 0.9999985 with only two spare units weighing 20 pounds total. This will meet the mission requirements satisfactorily. However, since the function and resulting data is temporarily lost, other forms of improvement may be worth considering. In order for the CTE to meet the reliability goals for the frequency divider outputs (0.999999), each of the networks must have a reliability in excess of 0.999999; considering a 2 out of 3 network. The required network failure rate is therefore:

$$0.999999 = 1 - 3\ (\lambda t)^2$$

$$1 \times 10^{-6} = 3\lambda^2$$

$$1 \times 10^{-6} = 3\lambda^2\ (1.68 \times 10^4)2$$

$$= 3\lambda^2\ (2.282 \times 10^8)$$

$$\lambda 2 = \frac{1 \times 10^{-6}}{8.46 \times 10^8} = 0.0011 \times 10^{-14}$$

$$\lambda = 0.33 \times 10^{-8}$$

This represents the required failure rate of one leg of a typical frequency divider network considering a mission time of 700 days. For comparison, the one leg of a typical frequency divider network in the present system has a failure rate of 0.121×10^{-6}. This comparison shows that approximately 2 orders of magnitude failure rate improvement is required for a frequency divider network. Data indicates that this may not be feasible within the next decade.

Because of the character of the CTE design, it may be maintained at three levels; cards, modules, or parts. Of the three alternatives, the part level is entirely impractical, and the module level is possible but difficult in the present design. However, the card level can be replaced fairly easily, even in its present form, see Figure 4.77. Sparing at that level and providing the necessary access through the case will improve the contribution to P_s to 0.999993 with only four spare cards as indicated in Table 4.50. This added only 8.3 pounds to the spares complement and the mission objectives can be met without compromise.

Table 4.50. CTE Card Level Sparing Analysis

Card	Reliability	Spares Required	Contribution to P_s	Spares Weight in lbs.
Frequency Divider	0.997	1	0.999996	1.75
Time Accumulation				
Type A	0.98	1	0.9999985	1.6
Type B		1		1.6
Power Supply (2 per card)	0.9989	1	0.999999	3.3
Totals	0.978	4	0.999993	8.25

4.7.4 An Optimized Maintenance Concept

If it were considered desirable to keep the timing and frequency division functions available at all times, other forms of M and R should be considered. In this case, M and R must be accomplished so that the signals are not interrupted — the inherent functional redundancy continues to operate properly while the failed element is replaced.

In order for the CTE to remain operating during repair activities, the packaging design must complement the electronic design for redundancy. That is, packaging must be similar to the configuration shown in the

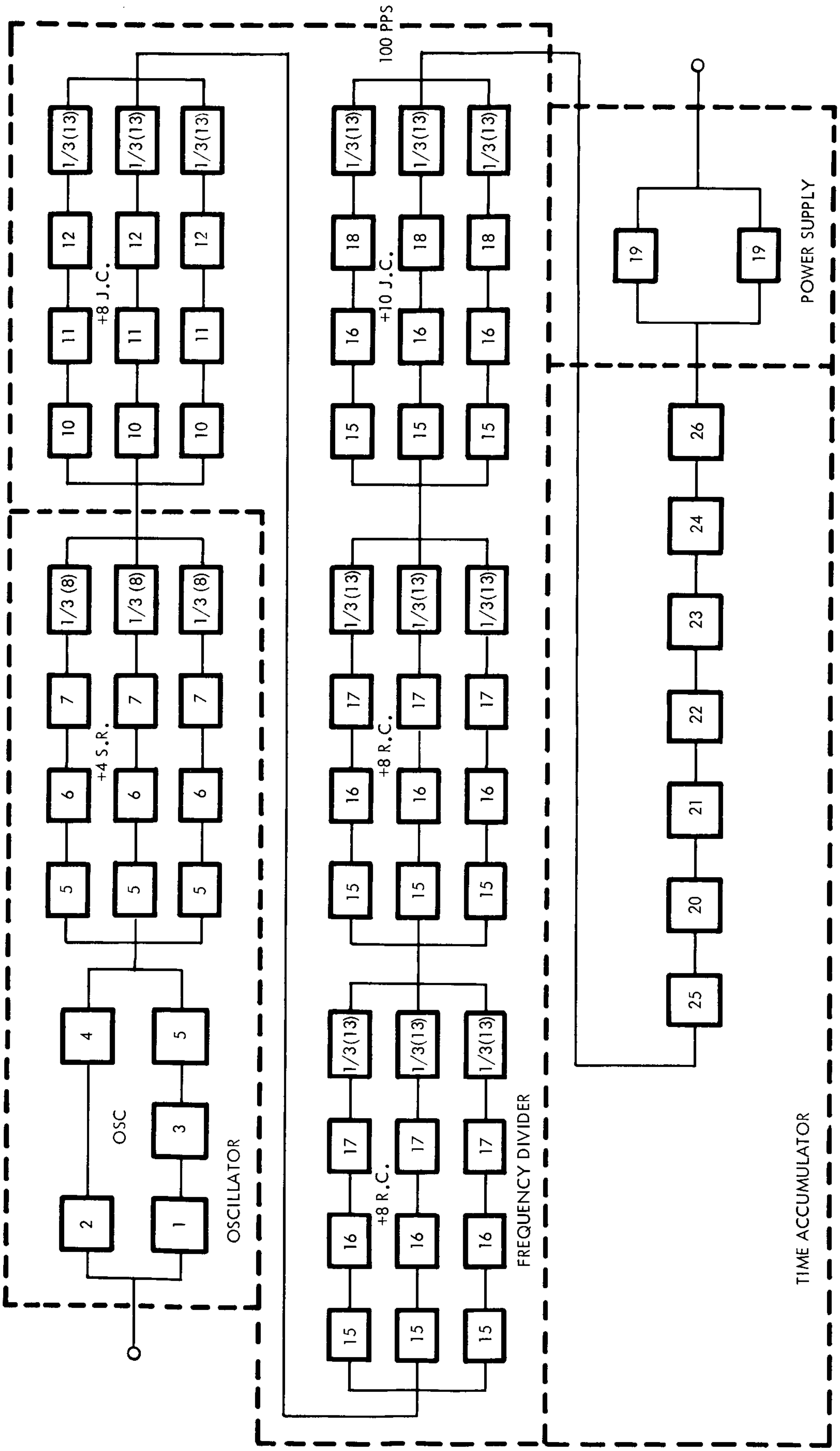
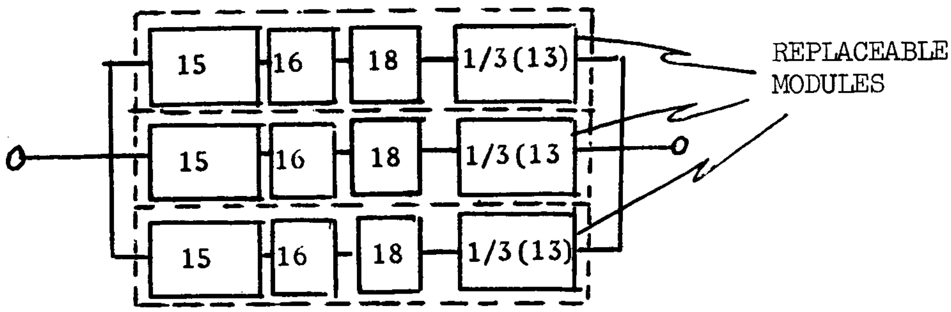


Figure 4.77. Parallel Time Code Otuput, Reliability Block Diagram

reliability block diagram of Figure 4.76. For example, the following diagram shows the manner in which this would be accomplished. Using a ÷ 10 Johnson Counter as a model:



In order to decrease the number of replacement items, circuits within a particular leg of a network may be combined provided the principle of "repair while operating" is not violated.

Repairs will be accomplished by module replacement. Consequently, no individual component parts need be carried aboard for maintenance of this system.

In order that the probabilities of success calculated in the previous section be realized during the actual mission, methods for quick recognition of a failed condition must be devised. Repair may be subdivided into elements of recognition, isolation, replacement and checkout. By use of modular replacement, time to replace and checkout will be short. The only uncontrolled time elements are those of recognition and isolation. These "allowed" repair times range between 2 and 109 hours. For any particular network, the recognition time must be much shorter. Since the "allowed" repair times are much shorter than a normal weekly or monthly preventive maintenance period, a system of network monitoring become necessary. In its simplest form, this system consists of visual indicators showing the conditions of each leg of a redundant network. Such a system would eliminate the need for sophisticated built-in test equipment.

Preventive maintenance would still be required. However, these activities could be minimized and would be performed on a scheduled basis to detect degraded conditions indicative of impending failure.

To accomplish the approach described here, certain modifications to the physical configuration become necessary. The most prominent of these are listed below:

1. **Repackaging of the unit so that maintenance philosophy may be accomplished.**

2. Provisions for plug-in modules.

3. Inclusion of additional test-points for preventive maintenance.

4. Provision for the monitoring system described previously.

To accommodate this design concept for maintainability, the modules presently existing need to be subdivided and repackaged in a form representing the redundant electrical design. This includes the redesign of the various p. c. cards to accept plug-in modules. In addition, the CTE case must now incorporate provision for quick access to the modules. The printed circuit cards would require corrugated or flat cabling allowing them to be withdrawn from the case while operating. The size and weight of the CTE would increase by about a factor of two as a result of these changes.

It should be reemphasized that this concept is not required. Rather, it is an objective for the optimized design.

4.7.5 Conclusions and Recommendations on the CTE

The conclusions to be drawn from these analyses indicate that the CTE is capable of operating for the long durations imposed by planetary missions and that mission requirements can be most feasibly accomplished by providing for some in-flight maintenance. Such an approach would insure an acceptably high probability of mission success.

Maintenance provision at the card level seems to be the most desirable approach since minimum modifications and maintenance complexity results. Any temporary loss of timing can be compensated for by use of a standby unit and/or the Up Data Link.

The use of a proven, qualified design for extended mission application is not without merit. The use of such hardware as a working baseline shortens considerably the design, development, production cycle. In addition, greater confidence can be placed in the operability of the resulting end product.

V. CRITICALITY II SUBSYSTEMS

In defining the mission success and safety objective and as a ground rule for this study, the system functions that did not directly affect crew survival were separated by criticality rating from those that did. They are the Criticality I grouping. A further separation involved the remaining functions – system functions that were related to crew safety and well being, but that were not essential in that there was a way home. These were to be included in the availability assessment and are considered Criticality II functions. Thus, such functions as artificial gravity and communications (excluding of data link) are separated into this category. Grouping system functions under Criticality II is a somewhat arbitrary way of identifying functions that do not require quite as much emphasis as crew sensitive functions. They may be called "Crew Comfort systems" in that they may affect the phisiological or psycological condition to some minor extent. These are assessed in the following section.

5.1 SPINUP AND DESPIN AND PRECESS CONTROL

(Note: Much of the data used in this section was provided by the Honeywell Corporation through Reference 4.4.)

5.1.1 Functional Description

The spinup and despin and precess control functions are required because of introduction of the artificial gravity mode. The control system must perform four major functions:

1. Monitor artificial gravity status
2. Control artificial gravity acquisition
3. Control precession of spin phase
4. Wabble damping

Of these functions, only the first two introduce any requirement for automated control. Precession can be accomplished by manual activation of the precession engine when, and if required. Wabble damping can best be controlled through a passive system using a viscous column of fluid as the control medium. For details see Reference 1.1.

Monitoring artificial gravity status is a simple function involving the sensing of the achieved gravity level and displaying the resultant to the crew. Its operating time is 15,727 hours for the baseline mission, the total of all the artificial gravity phases.

The control of artificial gravity acquisition involves control of the separation and extension system and control of the spinup and despin engines. The sequence of operations involves, first, the rotation of the spacecraft at about one rpm in the plane of the intended spin; next, the separation latches are released and the extension system activated, permitting the two sections of the spacecraft to separate on cables, which are kept at constant tension; then the spinup engines are activated at the selected radius, to bring the rotation rate to within the comfort zone.

Spinup and Despin Control

The spin vector direction and magnitude and the g level in the MM during spinup and despin operations will be sensed by rate gyros (BMAG's) and an x-axis accelerometer. These will be displayed in the MM on FDAI(s), or equivalent, and a g indicator. Other displays for the spin and despin operations should include a cable length meter and warning indicators of cable winch stall, or winch malfunctions as yet undefined.

Cable deployment will be accomplished by cable winches, each consisting of an electric motor, motor controls, gear box, reel-and-cable laying guide, deployed cable length and rate sensors, brake, and structure. A passive reel concept employing spring tension to control cable reel-in and reel-out may be worthy of consideration.

Spinup torques will be supplied by reaction jets on the MM; Apollo RCS engines will perform this function. Even if an additional RCS is required for spinup (SU) and despin (DS) torques, the normal MM RCS can be used for MM attitude control during SU/DS operations.

Cable deployment, spin torque, and MM attitude will be controlled by the MM G&N computer during SU/DS operations. Measurement and control of total spacecraft velocity change during SU/DS operations requires that the MM G&C system have, in effect, the capability of executing small midcourse corrections while spinning.

Spinup and despin operations have been estimated to require from 40 to 70 minutes each. Four to six SU or DS operations will be required for each leg of the mission: initial spinup following planet departure and perhaps after an initial correction; two for midcourse nonspinning periods for navigation and/or correction; and terminal despin during planetary approach. With a maximum of four mission legs (triple planet flyby), the total time for spinup and despin operations should be 24 hours or less per mission. The cable winches and spinup reaction jets must of course remain functional for these operations after two years of exposure to the space environment.

Spin Vector Control

During spinning mode cruise, the spin vector direction will be monitored and corrected as required. The principal constraints on spin vector orientation are expected to be spacecraft thermal control, which will probably require that the spacecraft-Sun line of sight be approximately normal to the spin vector, and field of view requirements for the earth communications antenna or other sensors.

The spin vector direction can be partially determined from the spinning spacecraft by an array of four Sun sensors with a slit field of view spaced at 90 degree intervals about the nominal spacecraft spin axis. Differencing the elevation angles measured by opposite sensors, with suitable filtering of measurement noise due to MM wobble motions, will determine the spin vector deviation from the nominal spin axis, and averaging the four readings will give the spin axis-Sun angle. The time for one revolution determines the spin rate. Failure of one Sun sensor would still allow attitude determination with modified computer logic.

The complete three-dimensional spin vector orientation could be determined fairly precisely with an electronic-image-sensing, two-gimbal, star tracker, mounted externally on a despun base and used in conjunction with the Sun sensor array. (Star acquisition and image data processing with tracker platform wobble would present potential problems.) However, no real need is seen for this additional complexity. Present estimates are that the spacecraft spin orientation will not randomly change more than a few hundredths of a degree per day, which is less than the beamwidth of the communications antenna. If this is so, then, for earth communications periods a day or so apart, earth acquisition does not present a problem. The initial operation would be to true up the MM attitude, if necessary, so that the antenna despin axis would be nearly colinear with the spacecraft spin vector, as determined by the Sun sensor array. This would be accomplished by RCS and perhaps slight cable repositioning with the cable winches. The earth line-of-sight direction, in terms of angle with the spin axis and azimuth angle measured from the spacecraft-Sun line of sight would be computed, assuming that the spin vector had not been rotated about the Sun line since the last communications period. The antenna would then be pointed in the computed direction. Equalization of the antenna despin rate with the spacecraft spin rate, to provide stabilized antenna pointing, might be facilitated with a slit field Sun sensor on the antenna platform. This would require that the antenna itself have complete azimuth freedom relative to the despun platform on which the Sun sensor was mounted. This would require an extra movable joint in the system. Once the antenna received a signal from earth, it would track the signal, and the antenna pointing angles would be fed into the computer to update the spin vector direction.

Spin vector corrections would be controlled by the G&N computer and executed by the RCS. It is expected that corrections would be required only infrequently, with a very small RCS fuel requirement, but detailed studies have not been made.

If the spacecraft spin vector precesses only a small fraction of a degree per day, there is no need for continual monitoring of the spin vector, and operation of the Sun sensor array and relevant computer module could be limited to periods of antenna or scientific sensor pointing during spin mode cruise.

Wobble Damping

Wobble motions of the MM will result from coning of the spacecraft spin vector, resulting in erratic motion of the module center of mass, and from attitude excursions of the module, which may be made worse by the flexible cable system linking the MM to the EEM/Propulsion module. Normally these motions will be small, and will be damped by a passive viscous fluid damper. Active wobble damping by the propulsion module RCS may be required during spinup and despin operations and may also be required for certain mission events and contingencies. Studies of control logic for wobble damping of the artificial gravity spacecraft configurations indicate that adequate control will require 3-axis sensing of attitude deviation, angular rates, and accelerations. Thus, for short periods, the complete A&SCS system of strapdown gyros and accelerometers may be required. During these periods, control logic and RCS commands will be generated by a G&N computer module.

5.1.2 Reliability Analysis

The reliability logic diagram for the artificial gravity mode function is presented in Figure 5.1, an analysis of each function follows.

SU/DS Control Function Reliability

The SCS components required to control the gravity force are the ECA, AE/GD, driver amplifiers, and the rotation control (Figure 5.1). Because of the relative short operating time for the SU/DS control function (24 hours) as compared to the time these same components must operate in the zero-gravity configuration, this function has little effect on the mission reliability. The reliability estimate without provisions for maintenance and repair exceeds 0.999. Since some of the components herein are interchangeable with those in the A&SCS function, the probability of not having a functional gravity control function is less than one chance in 10^5.

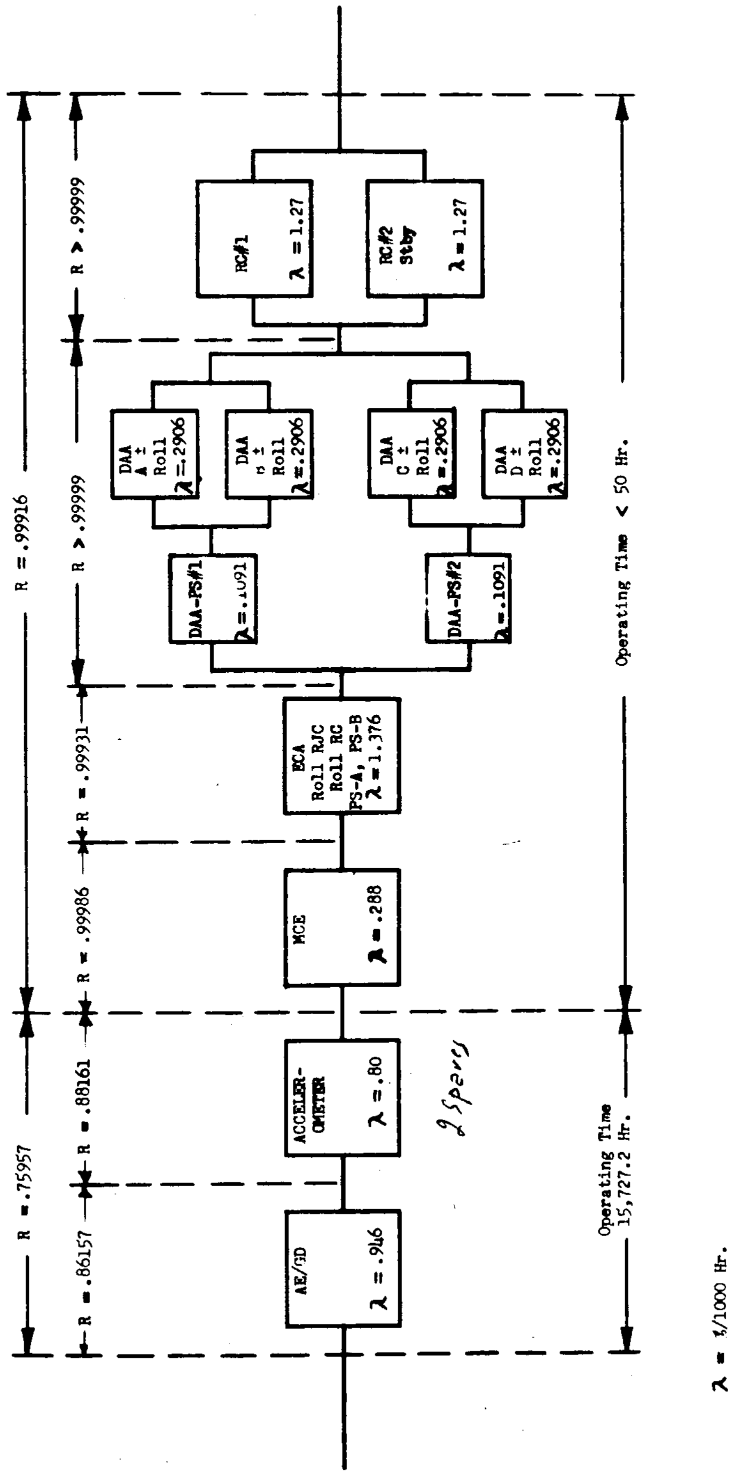


Figure 5. 1. Reliability Block Diagram, Artificial Gravity Control

Gravity-Monitor Function Reliability

The gravity monitor function employs two components not common to the Apollo Crew Module A&SCS. These components are the accelerometer used for sensing the artificial-gravity force and the display used to present the magnitude of this force. These type of components is used elsewhere in Apollo/Saturn and these data were used in this analysis. It has been assumed that knowledge of the gravity force is essential at all times while the vehicle is in the gravity configuration. The estimated reliability of the gravity-monitor function is 0.75957, based on 15,727 hours of operation. The chance of failure of both components of this function is greater than one in ten; therefore, both require some improvement in their reliabilities.

5.1.3 Availability Analysis

Inspection of the operating times and failure rates in the gravity-configuration reliability logic diagram (Figure 5.1) reveals the weak links are the gravity display and the accelerometer. Two spare gravity displays will raise the display reliability from 0.86157 to 0.999951. Two spare accelerometers will raise the accelerometer reliability from 0.88161 to 0.99970.

Considering these spares and the redundant ECA provided for a similar function in the A&SCS system, the reliability of the gravity-configuration is raised to at least 0.99907. This is sufficient to exceed the desired mission goal and is achievable with contemporary Apollo/Saturn hardware. Replacement of these components as a whole seems the most practical approach since they are small and light and not easily disassembled.

5.1.4 Conclusions and Recommendations on Artificial Gravity System

The analysis reveals that the control of the artificial gravity mode can be done safely and reliably. Table 5.1 indicates that the addition of only four spares to contemporary systems will assure system performance with a more than 0.999 probability. No requirements in the control area tax the state of art. However, none of these functions has been performed on earth or in space, and for that reason a detailed study and test program should be initiated to demonstrate feasability and identify both the potential design and operational problems.

Table 5.1. Spares Requirements, Artificial Gravity Control

Component	Number Required	Weight (pounds)	Volume (cubic inches)
Accelerometer	2	5.2/10.4	73.0/146
Gravity display	2	8.9/17.8	350/700
Totals	4	28.2	846

5.2 COMMUNICATIONS AND DATA SYSTEMS

(Note: Much of the data used herein was provided by the Collins Radio Co. through Reference 5.1.)

5.2.1 Functional Requirements

Specific functional requirements for the baseline mission Communications and Data System (CDS) were identified in the baseline mission description in Reference 1.1 and 5.1.

The telecommunication and data management system defined by the study efforts consists of three major subsystems: mission module communications, probe-support communications, and the earth entry module telecommunications. Each subsystem contains several primary-function elements that are interconnected to provide a multiplicity of functions. Lifetime, crew safety, and mission success objectives indicate the need for multiple-redundant and backup modes of operation; each system is redundant in itself in several ways, and each communications subsystem will, as an alternate, backup the operation of the others.

(1) The Probe Support Communications Subsystems (PSCS) is an identical duplicate of the MMCS and may operate in parallel or as an alternate to the MMCS for all functions. When used for communications with probes, the system provides the following functional capabilities:

- Probe tracking and ranging from spacecraft
- Transmission to probes of spacecraft-originated commands
- Reception of probes digital telemetry
- Reception of probes video.

Since this is a Criticality III system, no detailed analysis is devoted to it. However, since the duplicate functions in the Mission Module (MMCS) are analyzed, the results apply directly to these systems functions.

(2) The EEM CDS functions are required for launch operations and earth orbital operations support and during planetary injection, velocity changes, and earth reentry. Requirements imposed on these functions are similar to Apollo in every respect, and the Apollo capabilities will more than meet the EEM mission functional requirements. A function block diagram of these requirements is presented in Figure 5.2.

both, of the remaining stations. Because of this feature, any two of the stations will fulfill all the functional requirements.

The Premodulation Processor contains the circuits required to provide for the interface connections between the spacecraft data gathering equipment and the radio frequency transmission equipment. It contains signal modulation and demodulation, signal mixing, and signal switching circuits to provide the required signal paths for a given mode of operation.

The Data Storage Equipment consists of an 11-channel, 3-speed, magnetic tape recorder and reproducer. In its present configuration, it has the capability to record and reproduce two channels of digital data and five channels of analog data. Record and reproduce electronics are not supplied for the remaining channels. The capability to add these channels is inherent in the design concept. This equipment comprises tape deck, record electronics, reproduce electronics, a signal conversion electronics, tape motion and end-of-tape sensors, speed detector, switching electronics, and magnetic tape.

The Pulse Code Modulation (PCM) Telemetry Equipment consists of a combination of functional units necessary to combine various analog and digital measurements into binary-coded digital signals. The PCM is capable of sampling all or portions of its input signals, depending on which of two sampling programs is remotely selected. This equipment makes extensive use of internal redundancy to improve its reliability.

The Unified S-Band Equipment (USBE) consists of two redundant S-band, phase-locked transponders and one FM transmitter. Each transponder consists of a transmitter, receiver, and power supply. The FM transmitter is electrically independent of the transponders. Only one transponder is active at a given time. However, simultaneous operation of the FM transmitter and one of the transponders is possible. The USBE will be used only in the sphere of influence of the earth at the beginning and end of the mission.

The S-Band Power Amplifier consists of two 20-watt, traveling-wave-tube, power amplifiers with associated power supplies, control circuits, and a multiplexer mounted on a common chassis. It provides for either amplified or bypass transmission in the PM mode. It also provides for amplified, but not bypassed, FM transmission. The multiplexer provides a receive signal from the antenna and isolates the amplifiers so that PM and FM transmission are possible simultaneously or separately. The PM bypass mode is construed to be for emergency use only. The EEM subsystem is independent of the mission module systems except for possible transfer of signals and data between the two modules.

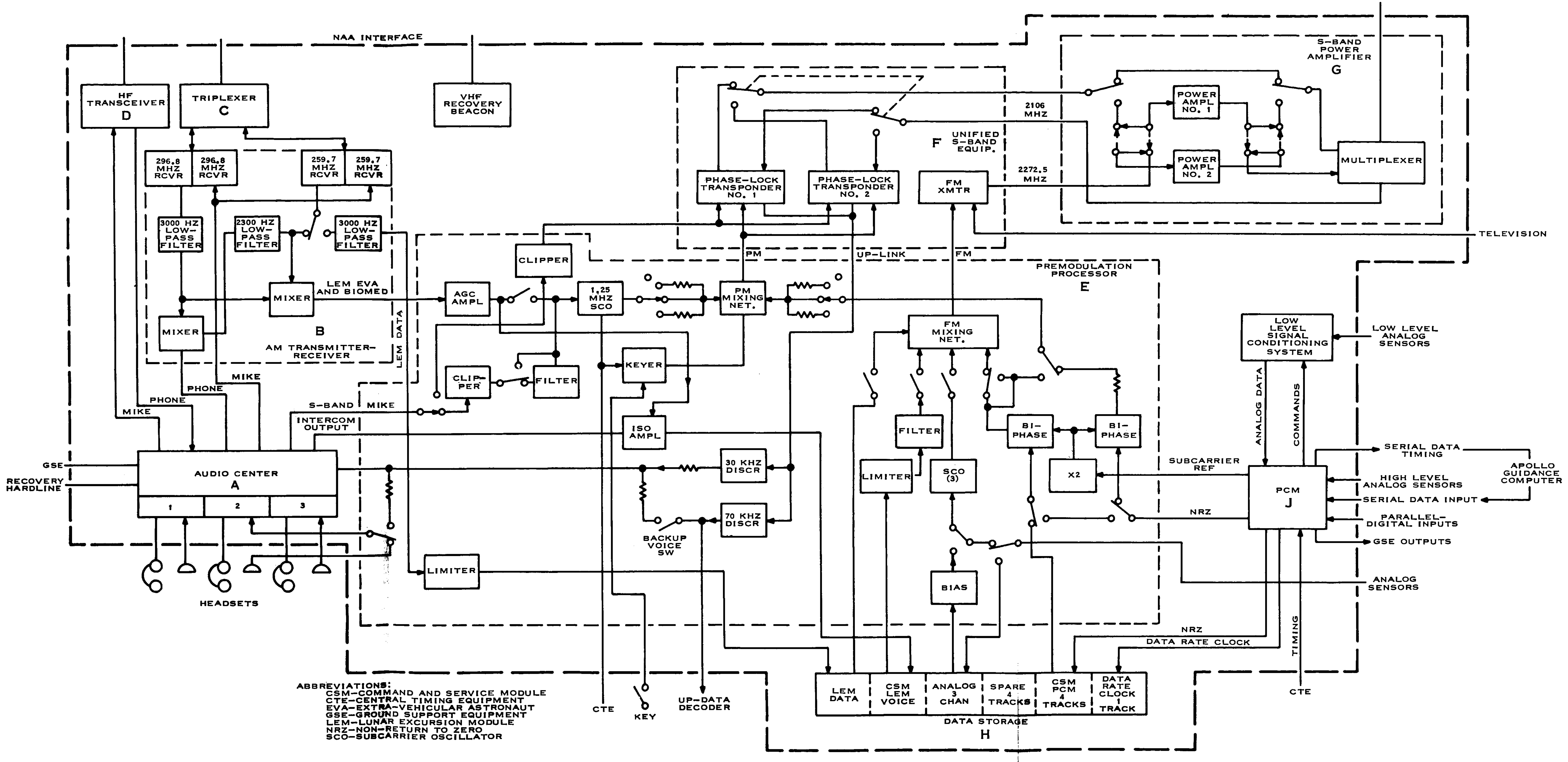


Figure 5.2. Earth Entry Module C&D Subsystem, Block Diagram

FOLDOUT FRAME 1

FOLDOUT FRAME 2

Several capabilities are included for use during earth orbital, EVA, and reentry/landing/recovery operations. The S-band equipment that normally supports the lunar mission during the non-earth-orbit phases of the lunar mission is not required as such for the EEM. It is, however, used in the MM. Other equipments may be used during all phases of the mission. A brief discussion of each piece of equipment follows.

High Frequency Transceiver comprises an HF transmitter, receiver, and power amplifier. It provides beyond-line-of-sight direction finding and voice communications during orbital flight and postlanding phases of the mission.

The VHF Beacon has the single function to provide line-of-sight beacon transmission during earth landing and recovery operations. It consists of a VHF amplitude modulated transmitter.

The VHF triplexer is a 3-channel, passive-filter device. It provides the necessary isolation to permit single-channel or simultaneous 2- or 3-channel transmission and/or reception with a common antenna.

The VHF AM Transmitter-Receiver comprises two independent VHF/AM transmitters and two independent VHF/AM receivers in a single package. These provide the necessary communications while in earth orbit. One transmitter and receiver provide for transmission and reception of voice communications on a single preassigned frequency. The other transmitter and receiver provide for transmission and reception of voice communications on a second preassigned frequency. This channel also provides for the reception of data (such as EVA biomedical) at the same frequency. External switching and the VHF triplexer permit the receivers and transmitters to be used in any combination of the two channels for voice communications.

The C-Band Transponder is used for near-earth tracking with ground-based radar. It consists of four superheterodyne receivers and a pulse transmitter and antenna switching network. These allow the transponder to receive and reply to any one of four antennas located in each quadrant around the EEM.

The Audio Center consists of four electrically identical sets of circuitry (stations), which enable parallel selection, isolation, gain control, and amplification of all spacecraft voice communications. Each station is operated from a remote control panel, which provides each astronaut with independent audio control of common S-Band, HF, WHF/AM, and intercommunication circuits. Each astronaut is provided with individual microphone and earphone jacks. In the event of a station failure, the failed-station microphone and earphones may be plugged into one, but not will fulfill all the functional requirements.

(3) The MM CDS functions are required to support the mission operations from post-injection to the time the crew enters the EEM for reentry. This amounts to about 16,800 hours of operation. However, all the CDS functions will not be required throughout that period and none is expected to be operated continuously.

Figure 5.3 presents the baseline CDS for the MM in functional form. It is designed to meet the requirements set forth in Reference 1.1 and Volume I of this report.

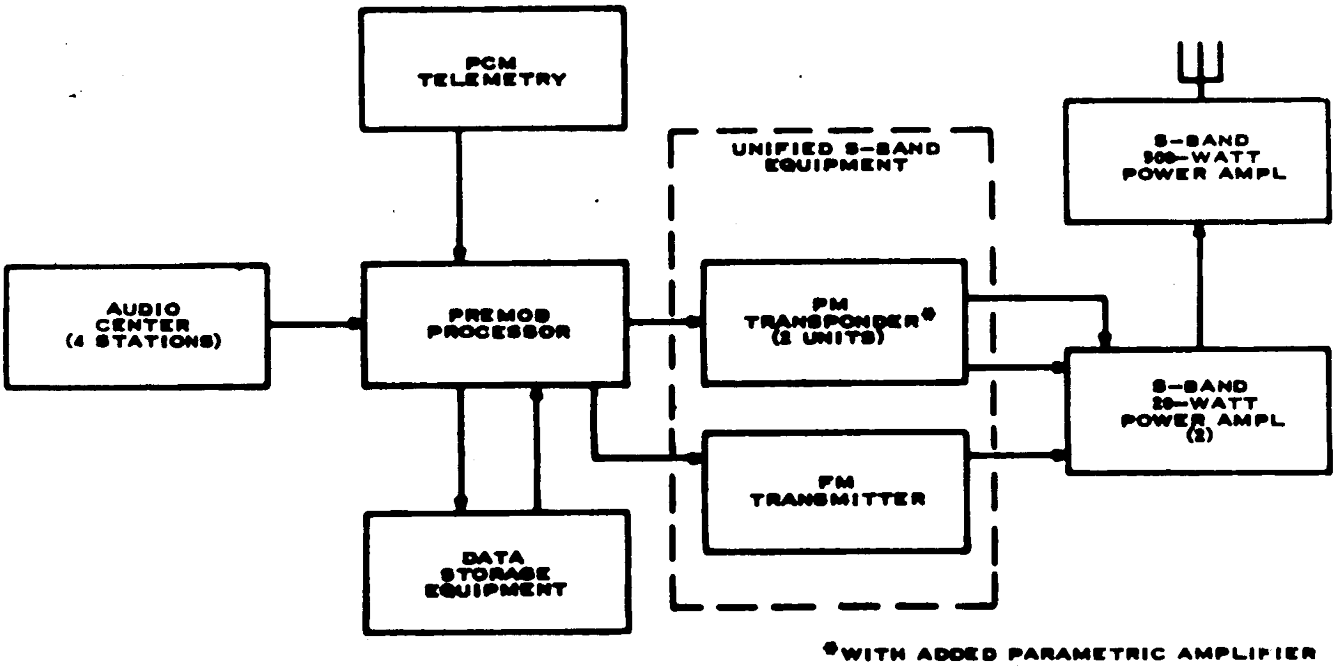


Figure 5.3. Baseline CDS for the Mission Module

The baseline MM CDS is considered to be made up of Apollo equipment as described in the previous paragraph but with the following important exceptions:

1. Low-noise, parametric amplifiers have been added to the USBE PM transponders.

2. The audio center has four instead of three stations.

3. A single 500-watt, S-band, power amplifier has been added.

The transmitter loop, operating at 2300 mHz, has dual 20-watt and dual 500-watt rf power amplifiers. The transmitter is 0.25 watts, PM or FM wideband, with one to three subcarriers as well as direct carrier modulation. Range-code and command-code generators and detectors and a variety of signal modulators and detectors are provided. Several phase lock and FM receivers are available. In general, a patch-panel approach would permit coupling functions as needed. The smaller equipment would have internal triple or quadredundant circuitry; larger equipment (power amplifiers) would be switchable pairs, supplemented by replaceable spares from storage.

Other characteristics of the system can be summarized as follows:

- Parametric receiver front ends with the noise figure of approximately 3 db.
- Variable data rates and bandwidth
- Low-gain backup antennas
- Coherent S-band transponders with a turnaround ranging channel
- Ranging and Doppler extraction circuitry
- Data and voice storage
- Audio centers and intercom.

The system is primarily intended to provide communications with earth, but can be used as a backup or a parallel capability for communications with probes. When used for spacecraft-earth communications, the system provides the following functional capabilities:

1. Transmission of spacecraft engineering and experiment telemetry
2. Relay to earth of probe telemetry
3. Transmission of spacecraft video
4. Relay to earth of probe video
5. Reception of earth-originated digital commands and encoded data
6. Reception and transmission of duplex voice
7. Transmission of emergency keying
8. Reception and retransmission of earth-originated spacecraft tracking and ranging signal.

Digital telemetry and command data are transmitted through PCM/PSK/PM. With a 210-foot-diameter ground antenna and 55 K ground-system noise temperature, telemetry can be transmitted at approximately a million bits per second at Mars encounter range and at 100 kbps at the range of 3.2 AU, with a circuit margin of 3 db. The corresponding command and encoded updata rates are 70 and 7 kbps using a 10-kw transmitter and an 85-foot-diameter antenna at ground. With a 210-foot ground antenna, the uplink rates are 400 kbps at Mars encounter and 40 kbps at 3.2 AU.

Wideband FM is used for the transmission of video data. Assuming a phase-lock FM demodulator at the ground and an output signal-to-noise ratio of 35 db, video images of commercial-television-type quality can be transmitted to earth at the rate of approximately two frames per second at Mars encounter range and 0.2 frame per second at 3.2 AU.

Duplex voice communications can be maintained throughout the mission except for periods of occultation by the Sun and target planet. Voice communications among crew members is provided by means of a hardwire intercom. For spacecraft tracking and ranging, a coherent carrier frequency translation and a turnaround pseudo-random code are used, phase modulated upon the carrier. The code bit rate of one megabit per second provides a range measurement accuracy of approximately 0.1 kilometer.

The main physical characteristics of the system are estimated to be as follows: weight, 700 pounds; volume (excluding 19-foot-diameter antenna), 6000 cubic inches; power consumption, up to 3200 watts.

5.2.2 CDS Reliability Analysis

(1) The EEM CDS reliability estimates are based on the data in Table 5.2 and the duty cycle estimates established for the baseline mission in Volume I of this report. The EEM CDS will display a reliability in excess of 0.99 and require no spares.

In the EEM subsystem evaluation, no consideration was made for the use of this equipment to back up the mission module subsystem or vice versa. The EEM CDS can be considered a backup to the mission module systems, and, conversely, both the PSCS and MM CDS can be used to compensate for defective EEM systems. It seems unlikely that portions of the EEM CDS would be used except when manned for either a normal or emergency return flight operation, and these will probably not exceed 100 hours.

(2) The MM CDS reliability estimates are based on the data presented in Table 5.4, which was, in turn, derived from demonstrated Apollo data, except those given for the 500-watt S-band power amplifier. These estimated values are based on experience with the 20-watt traveling-wave tubes (TWT) amplifiers and part vendors' expected characteristics of high-power TWT's.

Table 5.2. Apollo Block II C and D Subsystem Data

Primary Function	Equipment	Failure Rate (%/1000 hr)	Weight (lb)
Recovery	HF transceiver	3.67	6.1
	VHF beacon	0.109	2.3
EVA and near earth	Triplexer (Note 1)	0.0067	1.6
	VHF AM transmitter-receiver (Note 1)	1.519	12.2
	C-band radar	6.50	24.5
Translunar, lunar, and transearth	Audio center (3 stations) (Note 2)	0.51	7.5
	Premodulation processor (Note 2)	0.665	11.4
	Data storage equipment (Note 2)	4.50	40
	PCM (Note 2)	0.111 (Note 3)	42.7
	USBE PM transponder	1.94	31.6 (Note 4)
	FM transmitter	0.68	
	20-watt S-band power amplifier	1.21	31.6 (Note 5)

NOTES:
1. Also used for lunar communication between astronauts.
2. Also used in other phases of the mission.
3. Failure rate reflects built-in redundancy.
4. Includes two PM transponders and one FM transmitter.
5. Includes two redundant amplifiers.

Table 5.3. EEM Subsystem Reliability Estimates

Assembly	Mission Operating Time (hr)	Probability of Mission Success	Comments
Audio center	109.5	0.99926	Four stations.
VHF/AM transceiver	45.0	0.99984	Includes both EEM-MSFN and EEM-EVA links.
VHF triplexer	45.5	0.99999	
HF transceiver	13.5	0.99996	Assume same duty cycles as Apollo
Premodulation processor	49.0	0.99907	
Unified S-band equipment	1.5 receive 11.5 transmit	0.99991	
S-band power amplifier	11.5	0.999999	
Data storage equipment	35.0	0.99840	
PCM	61.0	0.99993	
VHF recovery beacon	82.0	0.99991	
C-band transponder	93.0	0.99397	
EEM CDS		0.99	

Table 5.4. Mission Module Baseline Systems Equipment

Equipment	Failure Rate (%/1,000 hr)	Weight (lb)
Data storage equipment	4.50	40.0
USBE: PM transponder	1.99 (Note 1)	31.6 (Note 3)
FM transmitter	0.68	
PCM equipment	0.111 (Note 3)	42.7
Premodulation processor	0.655	11.4
Audio center (4 stations)	0.68	10.0
20-watt S-band power amplifier	1.21	31.6 (Note 4)
500-watt S-band power amplifier (Note 5)	5.00	325.0
Total System Weight		492.3

NOTES:

1. 0.05%/1,000 hours has been added to cover required low-noise, parametric amplifier.
2. Includes two PM transponders and one FM transmitter.
3. Failure rate reflects built-in redundancy.
4. Includes two redundant amplifiers.
5. Failure rates and weights estimated.

The reliability requirement for the MM CDS functions vary with the criticality of the involved function. Establishing these requirements involved defining critical assemblies at the subsystem level and then critical subassemblies at the assembly level. Critical assembly and subassembly definition was based on a reliability weak link and system criticality assessment.

A weak link was defined as an assembly or subassembly having the lowest probability of surviving without failure in the system or assembly reliability configuration. The selection of weak links was further weighed by their criticality assignment established in Volume I of this report and related to the mission objective.

Criticality was factored into the requirements analysis by interpreting the probability objectives, assigned to the criticality classes, in terms of mission failures. Thus Criticality II, having a probability objective of 0.95, would be expected to produce only half as many failures per mission as Criticality III equipment having an objective of 0.90. The analysis was conducted with Criticality II equipment being weighted twice as heavily as Criticality III equipment.

Table 5.5 contains the various reliability parameters used to accomplish the mission module communications and data subsystem requirements analysis.

The first step in the analysis involved calculating the mission reliability for each assembly of Table 5.5 as well as the number of expected failures for a mission. The number of expected failures for Criticality II equipment was then doubled in order that they could be properly weighed against Criticality III assemblies. Results of these calculations are contained in Table 5.6.

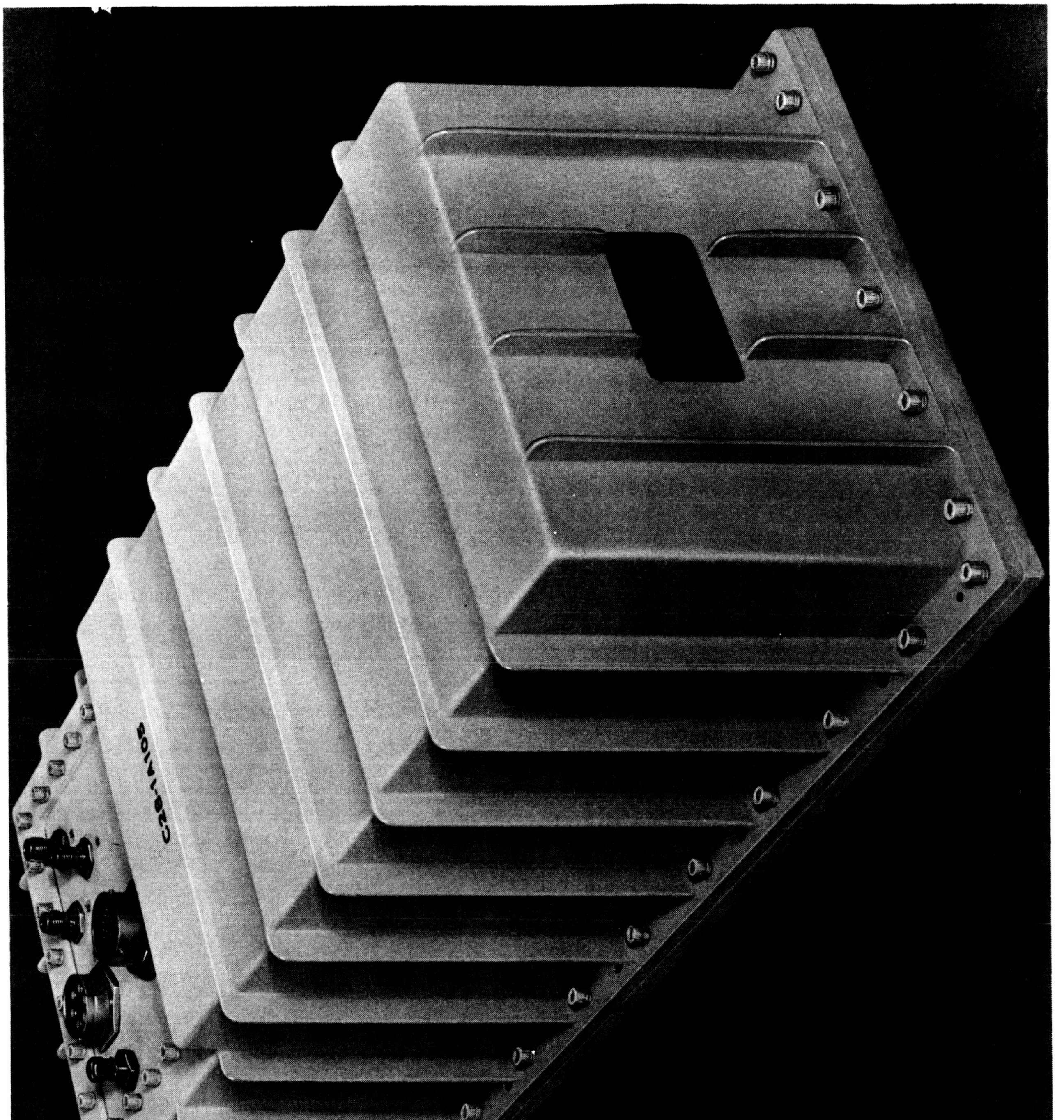
This analysis indicates that the USBE, data storage, and audio center all display a reliability far too low and results in a potential MM CDS reliability of about 0.379.

It will be noted in Table 5.6 that the USBE has the highest weighted number of expected failures and is therefore the critical assembly in the path. If the Apollo USBE had been used in the baseline system, the weighted number of failures would have been 0.343, which would have made the assembly the second critical assembly, still a prime candidate for improvement.

5.2.3 CDS Availability Requirements Analysis

The reliability analysis revealed several weak-link functions within the MM CDS that will not permit achievement of the mission objectives without some form of improvement. These are analyzed herein using the availability concept.

(1) The USBE displayed a potential reliability of only 0.7105 and since it is the major link from deep space to the earth complex, it should be in excess of 0.99. An examination of the present design concept revealed that the design could be maintained at two levels (Figures 5.4 and 5.5), either the major subassembly or the part level. Since this was the case and maintenance by assembly replacement was the optimum concept for improving availability, the spares indicated in Table 5.7 were found necessary to assure a P_s of greater than 0.99. The USBE required only 5 spare assemblies, amounting to only about 15 pounds.



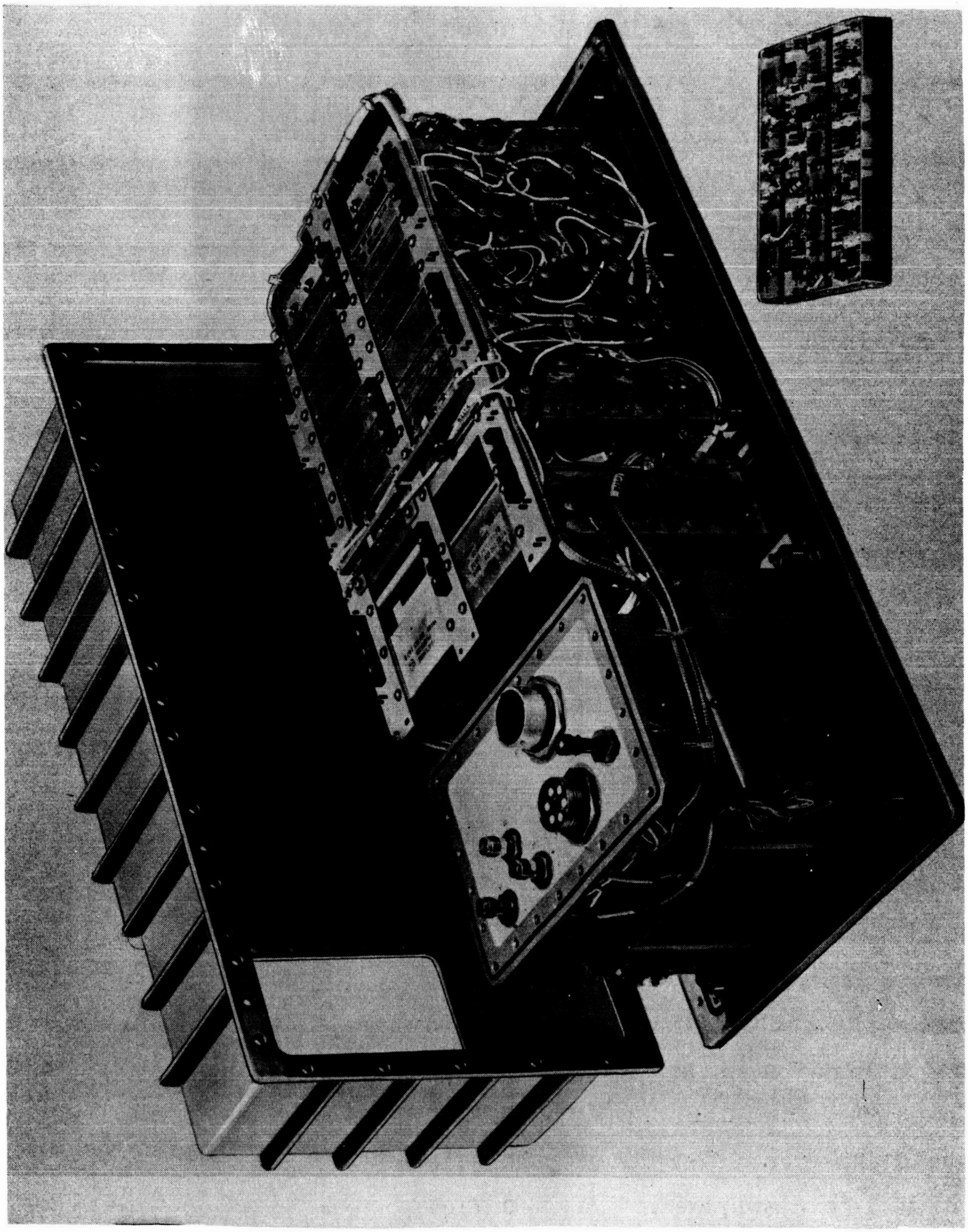


Figure 5.5. Exposed Modules, Unified S-Band Equipment

Table 5.5. Baseline Subsystem Reliability Parameters

Assembly Name	Failure Rate (%/1000 hr)	Operating Time (hours)	Criticality Class	Comments
Audio center	0.17	16,541	II	Failure rate is for each of four stations
Premodulation processor	0.655	5,851	II & III	Supports both communications and data functions
PCM	0.111	519	III	Failure rate is an average failure rate reflecting extensive use of circuit redundancy
Data storage equipment	4.50	8,510	III	
Unified S-band equipment:				
PM transponder	1.99	16,540	II	Failure rate is for a single transponder
FM transmitter	0.68	1,846	III	
20-watt S-band power amplifier	1.21	1,846	II	Failure rate is for each of two amplifiers considered in a standby redundant configuration
500-watt S-band amplifier	5.0	1,846	II	

Table 5.6. MM/CDS Reliability Requirements

Assembly Name	Probability of No Failure	Expected Failures (Weighted)
Audio center	0.8936	0.224
Premodulation processor	0.9624	0.076
PCM	0.9994	0.001
Data storage equipment	0.6821	0.383
S-band 20-w amplifier	0.9997	0.022
S-band 500-w amplifier	0.9118	0.184
Unified S-band equipment	0.7105	0.670
System	0.3789	-----

Table 5.7. USBE Requirements Analysis for Optimum Availability

USBE Subassembly	Probability of Success	Expected Failures (weighted)	Spares Required	Contribution to P_s
Receiver	0.85657	0.310	2	0.996
Power supply	0.90328	0.204	1	0.9950
PM transmitter	0.94469	0.114	1	0.9984
Rf tracking	0.99571	0.008		0.9957
Parametric amplifier	0.99077	0.016		0.991
FM transmitter	0.98745	0.012	1	0.9999
Miscellaneous	0.99681	0.006		
Total	0.7105	-	5	0.990

(2) The Data Storage Equipment (DSE) did not achieve the desired reliability goal at only 0.682. Since it is maintainable at the major assembly level (Figures 5.6 and 5.7), it was decided to select the required spares even though it was not required for crew comfort or safe return. It is in Criticality III and only associated with mission success (P_{ms}). Table 5.8 reflects the result of maintenance provision at the assembly level. The resulting contribution to P_{ms} was raised from 0.682 to more than 0.927 with only three spares, amounting to about 57 pounds. Repair to this component, which do not require supporting spares, can lower its contribution to P_{ms} to better than 0.999.

(3) The Audio Center (ACE) required improvement in its contribution to crew support requirements since its mission reliability for the four centers was only about 0.894. It was found that by adding one spare station as a complete component, the contribution to P_s was raised to over 0.994. The replacement problem was simplified considerably since the failure hazard at lower levels was evenly distributed and M&R at that level is possible but more complex. This is evident from the various views presented in Figure 4.8. Much is to be gained in terms of reliability and convenience with the spare station. It could be wired into the system and left off until needed. It would be available to all crewmen and controlled from its own panel, which would be placed in the most useful location. Another possibility would be to provide each station with its own connector on the ACE. A failure in one of the four operating stations would be corrected by moving the appropriate cable connector to the

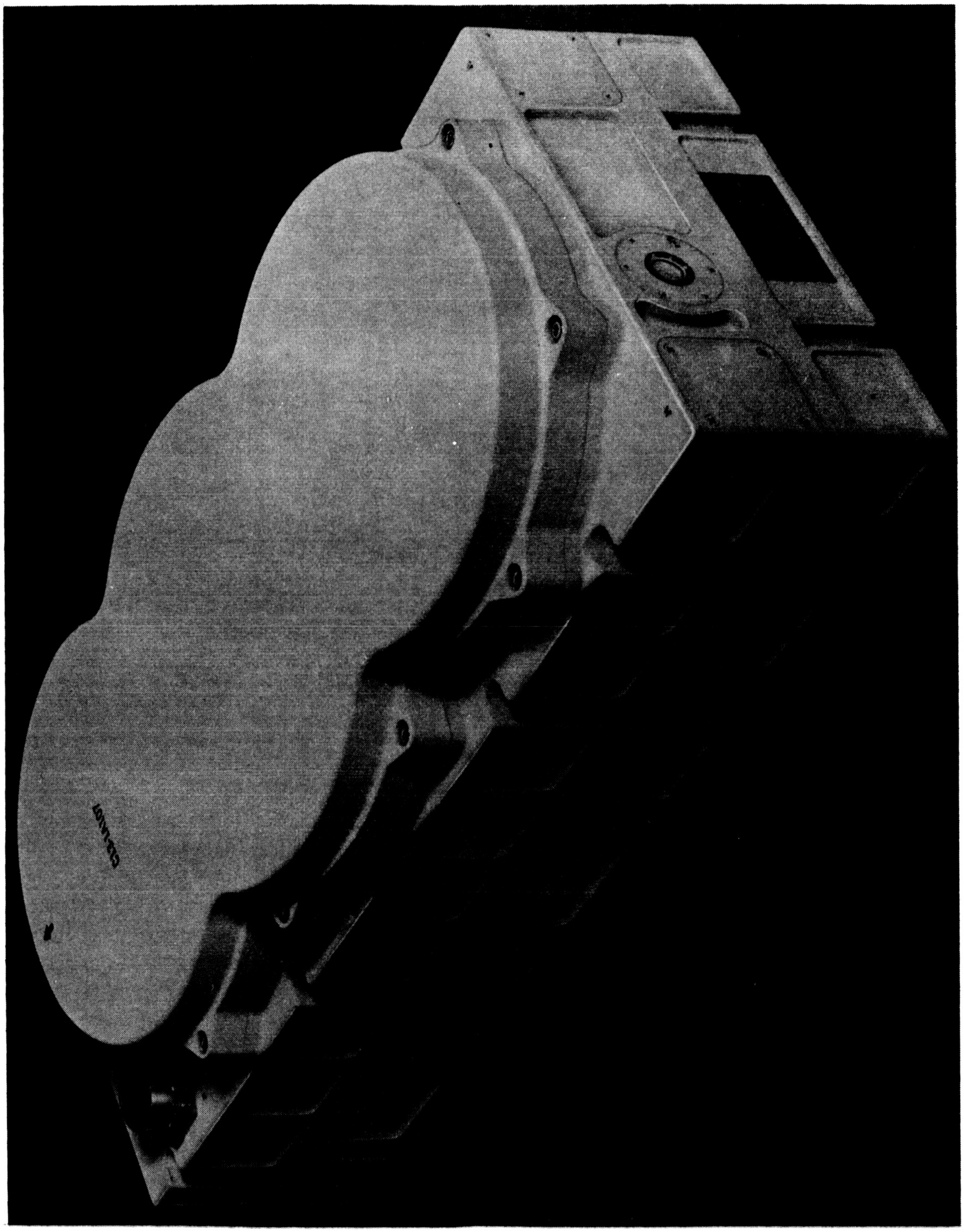


Figure 5. 6. Data Storage Equipment, External View

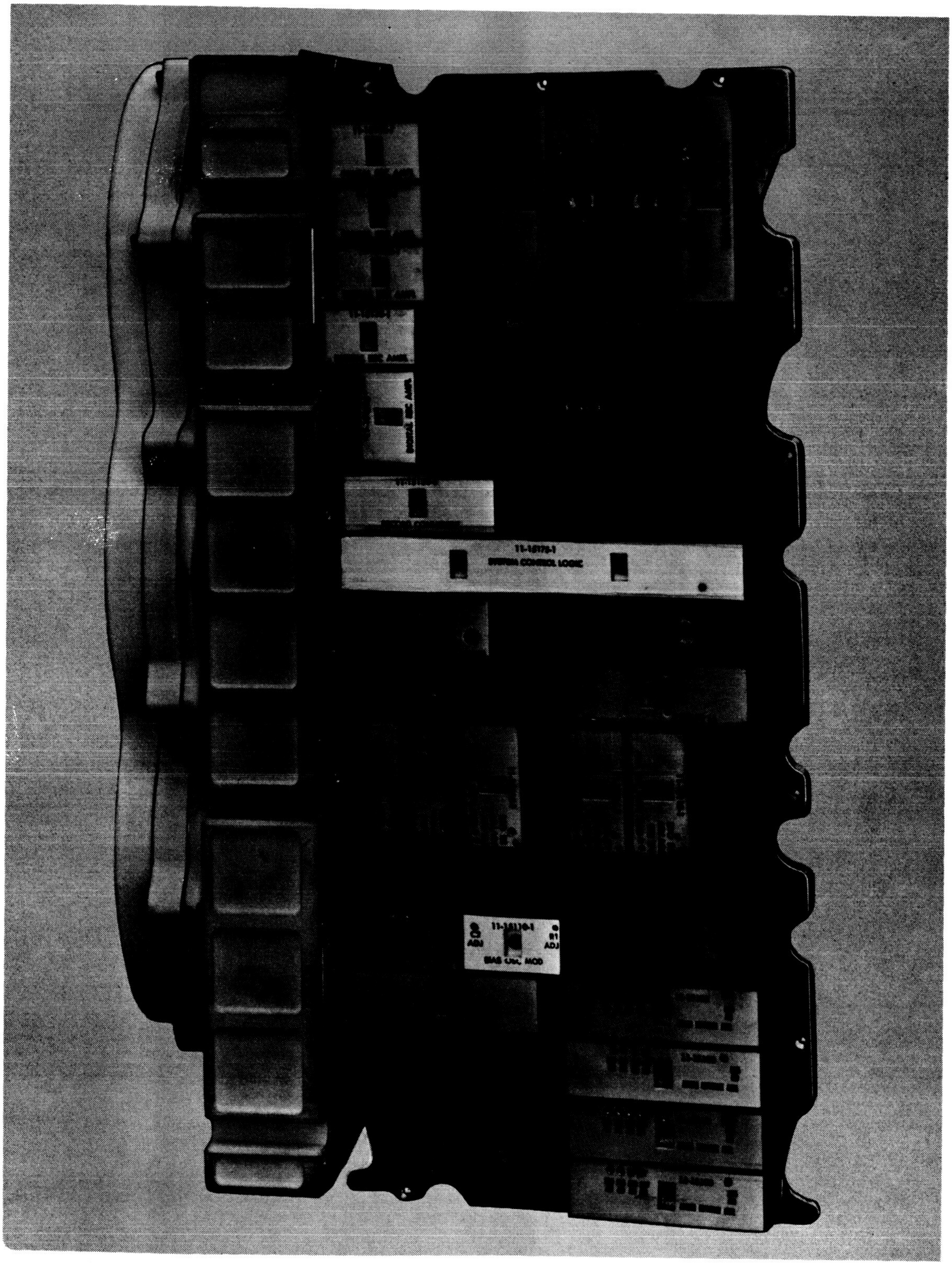


Figure 5. 7. Data Storage Equipment, Construction Exposed

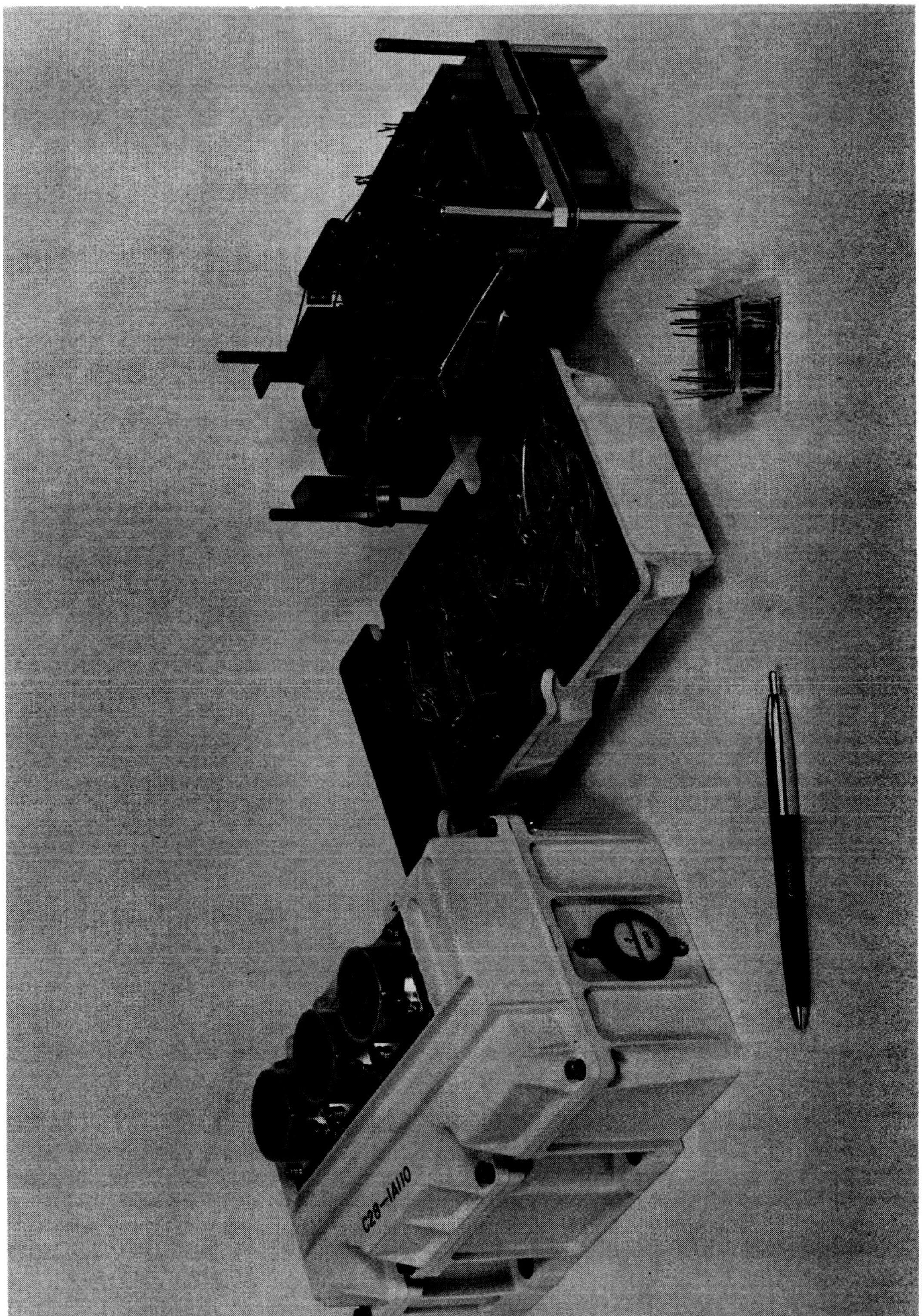


Figure 5.8. Audio Center Equipment

Table 5.8. DSE Requirements Analysis for Mission Success

DSE Subassembly	Probability of Success	Expected Failures (weighted)	Spares Required	Contribution to P_{ms}
Transport	0.7550	0.281	2	0.9970
Power supply	0.9704	0.030	1	0.9989
Control logic	0.9802	0.020	1	0.9998
Digital circuits	0.9792	0.021	1	0.9998
Analog circuits	0.9920	0.008	0	0.9920
Erase circuits	0.9930	0.007	0	0.9930
Heads	0.9940	0.006	0	0.9940
Miscellaneous	0.9900	0.010	0	0.9900
Totals	0.682	-	3	0.978

spare station connector on the ACE. Emergency station sharing, as used in Apollo, will still be provided.

(4) The 500-Watt S-Band Amplifier will probably present the most difficult problem in terms of maintainability. Its initial reliability for the mission is expected to be about 0.91, but since it is a new system function there is little applicable data to verify this assessment. It is, however, known that the traveling wave tube (TWT) will most likely contribute most of the failure hazard, and, for that reason, replacing the TWT is expected to be the optimum maintenance concept for this function. To raise the contribution to P_s from 0.91 to 0.995, one spare TWT is required, or a complete spare amplifier package will raise it to 0.9967. Two spare TWT's will raise it to 0.9996, and two amplifiers to 0.9999. The recommended approach is to provide at least two TWT's as spares because of the poor data base.

5.2.4 Maintainability Inferences on the CDS

Maintenance and repair (M and R) of the CDS was shown to be a requirement to achieve the MM system P_s objectives. The EEM system could achieve its objective without any maintenance plans. The resulting M and R requirement are reflected in the proposed final system configuration of Figure 5.9.

Maintainability features of the present designs, typical of contemporary hardware, are self-evident from the photos of Figure 5.10 and 5.11, the PCM telemetry and premodulation processor.

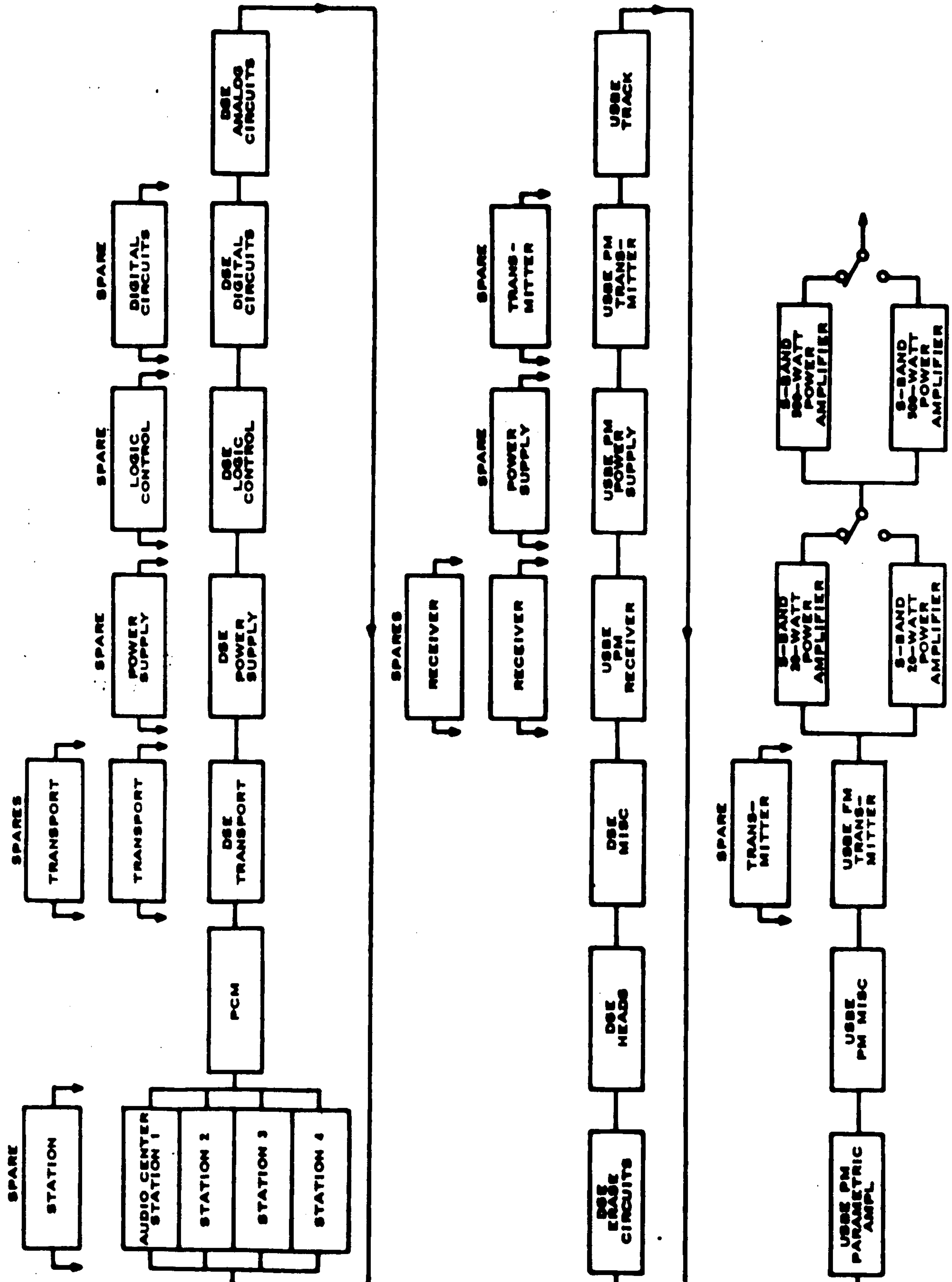


Figure 5.9. Final Subsystem Reliability Block Diagram

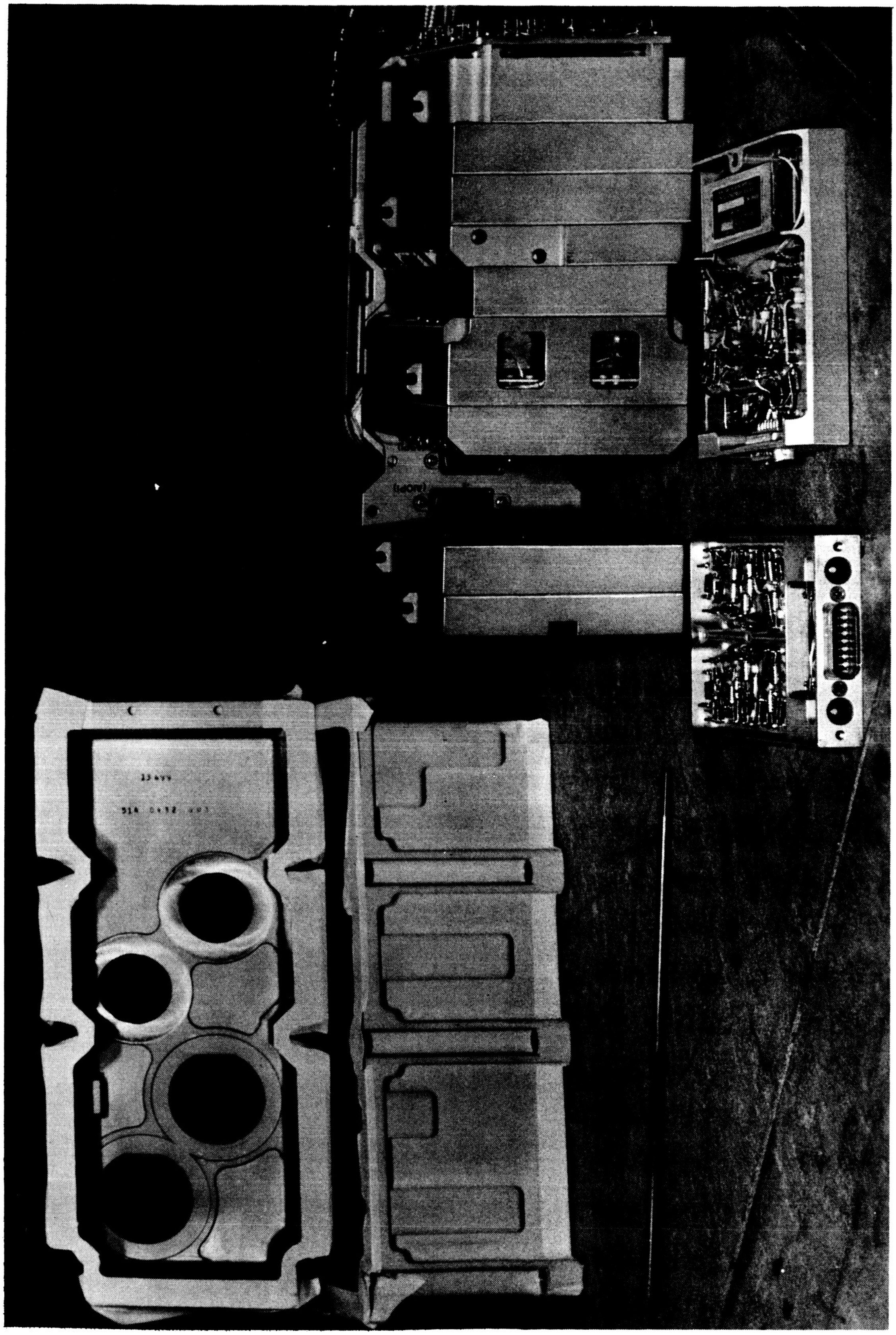


Figure 5.10. Premodulation Processor

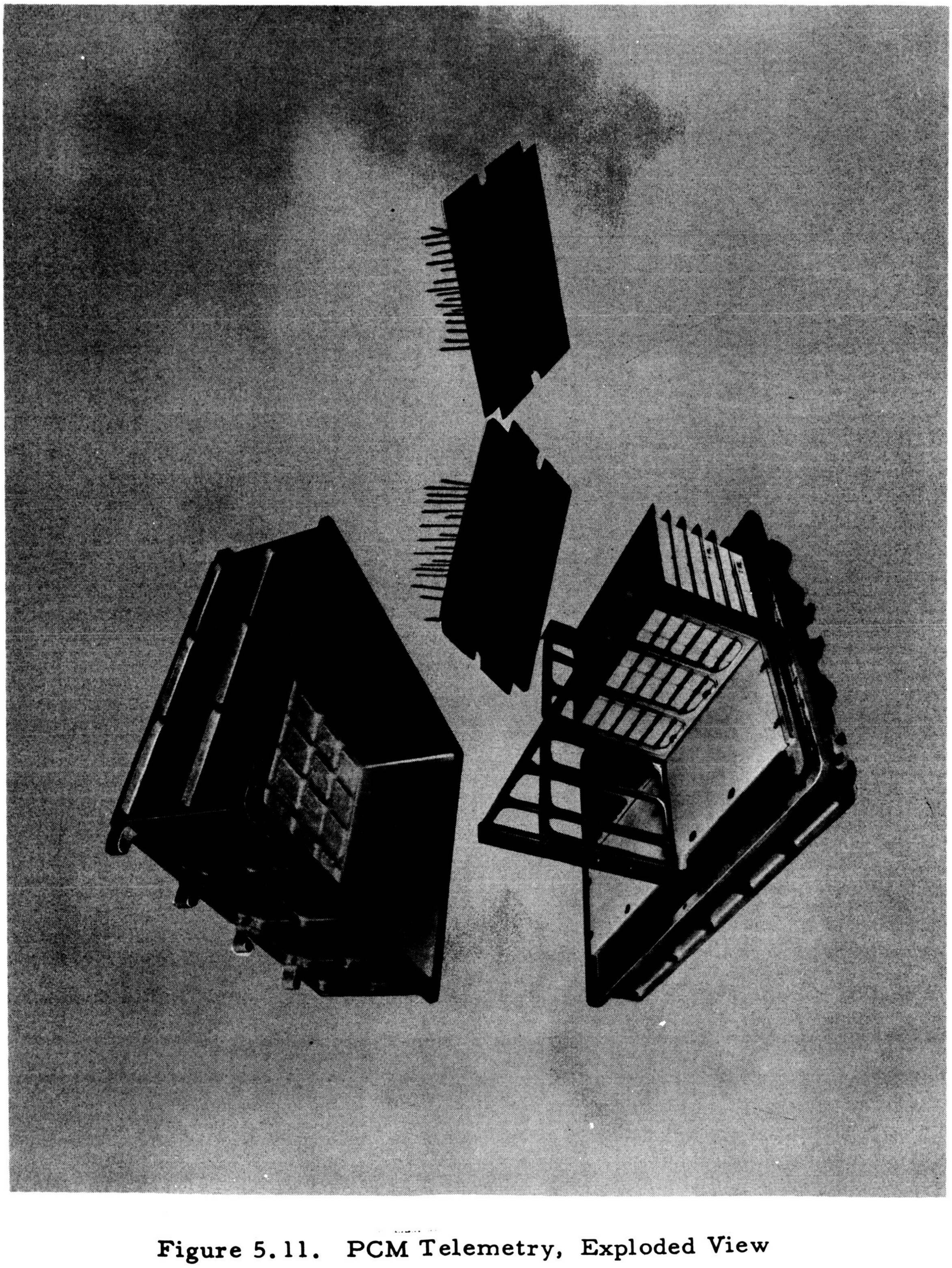


Figure 5.11. PCM Telemetry, Exploded View

Table 5.9. MM CDS Spares Complement

Equipment		Spares Assemblies	Number of Spares	Weight of Spares (lb)
DSE	0.967*	Transports	2	50.0*
		Power supply	1	2.5*
		Digital circuits	1	3.5*
		Control logic	1	1.0*
USBE	0.990			
		Receiver	2	6.0
		Power supply	1	5.0
		PM transmitter	1	2.0
		FM transmitter	1	2.0
PCM	0.999	None		
PMP	0.982	None		
Audio center	0.994			
		Audio station	1	2.0
20-watt S-band power amplifier		TWT	1	1.0
500-watt S-band power amplifier		TWT	2	3.0
Totals	0.98+	-	14	78.0

*Not required for crew comfort or safe return.

The following changes are required to the baseline system to arrive at a final system capability of meeting availability requirements:

1. The DSE requires changes to allow transport replacement as a maintenance action in space. Replacement of the power supply, digital circuits, and control logic is presently feasible, and some changes would improve capability.

2. The 20- and 500-watt amplifiers have to be designed to facilitate replacement of the TWT's.

3. The present method of securing covers should be modified to permit rapid access to replaceable modules.

5.2.5 Conclusions and Recommendations on the CDS

The study demonstrated that the CDS in the Apollo configuration could meet the mission objective with the addition of a 500-watt S-band amplifier and a low-noise-parametric amplifier to the USBE transporter. No modifications were required to facilitate the maintenance as recommended; however, some changes were desirable to decrease the spares complement required, and to permit expedient performance of the tasks. As Table 5.9 indicated, only 14 spares, amounting to 78 pounds, were required to meet all the CDS objectives. However, only 9 of these, amounting to only 21 pounds, were required for functions affecting crew comfort, and are really required to assure the planned operational mode.

The following are some recommendations resulting from this analysis:

1. A method of thorough subsystem checkout is needed aboard the spacecraft. This checkout can be used as a routine subsystem maintenance procedure such as checkout preceding critical mission phases. If the subsystem is known to be in operating condition just prior to a critical mission phase, the required probability of completing that phase becomes realizable.

2. Spare and redundant items should, in general, be at a functional assembly level, or a large high-failure-rate component such as a power amplifier tube. Simple modules should be used where possible. This would minimize repair time and maintenance-induced failures. It may also result in weight reduction by eliminating the need for sophisticated tools. Where this type packaging is not feasible (data storage equipment), it is recommended that complete equipment be carried as spares.

3. The relatively poor data storage equipment should be and can be made more reliable with a modest increase in weight. If the spares requirement were one instead of two units, a net overall weight reduction of 20 to 30 pounds would be possible.

4. Definite, well founded, failure rates for the periods of inactivity are lacking. These failure rates are assumed to be zero, and this is the normal practice.

5. Reliability figures for the 500-watt, S-band, power amplifier are estimates. The figures used in this study are believed attainable, but since this hardware has never been designed, built, and used, the analyses lacks validity. The design of such a function should start immediately because of the long development cycle required.

6. The data storage equipment is the only significant maintainability problem. This is because the present equipment does not lend itself to in-flight maintenance, other than replacing the entire unit. Two wired-in spares would appear impractical from a system wiring viewpoint.

7. Advanced packaging techniques recommended in Appendix IV would be desirable but not required. As a result the spares level would be reduced considerably.

5.3 ANTENNA SYSTEMS

Note: Much of the data used herein was provided by the Dalmo Victor Co. through References 5.2 and 5.3.

5.3.1 Functional Requirements

The antenna systems required to support the baseline mission fall into three groupings: the earth entry module antenna, the mission module to earth (MME) antenna, and the mission module to probe (MMP) antenna. The EEM antenna requirement will be satisfied by the Apollo antenna systems, excluding the deep space antenna. Since communications from the EEM will be limited to near earth distances, the omniantenna will satisfy this requirement. See Reference 5.3 for additional data.

1. The MMP antenna is required to establish and maintain communications with the planetary probes from the time they are launched at about planet perigee, minus 10 days, until earth takes over control of them at about + 10 degs. This is a noncritical (Criticality III) function. The communication range will include distances up to that encountered on the Apollo mission, and, for that reason, the Apollo deep space antenna (ADSA) will perform the required functions. The antenna is expected to have about a 28.6-db gain and a beamwidth of about 6 degree. The beamwidth will be unsatisfactory for monitoring and controlling all the probes at the same time and will therefore have to be operated in a modified scanning mode, time sharing the antenna between probes in a manner not yet defined. This antenna will also provide a backup capability to the earth link in case of failure. A picture of the antenna is presented in Figure 5.12. From it the inherent redundancy within the design proper may be seen.

The scanning is accomplished through a two-axis gimbal that requires no power during standby or launch and is locked in any position by removing the electrical power. The servo system is a closed-loop system which positions the gimbal anywhere between the ±6.6 degree limits when commanded. The system has been demonstrated reliable on several programs.

2. The MME antenna is required to maintain communications with the earth complex while the spacecraft is in both the zero and artificial-gravity modes. It will require a 19-ft. dish providing about a 37.2-db gain and about 1.5 degrees beam width. With minor modifications, the Apollo antenna gimbal will provide the mounting requirements. This has been demonstrated by Dalmo Victor, (Reference 5.3). This system therefore provided the data for the MME antenna analysis herein.

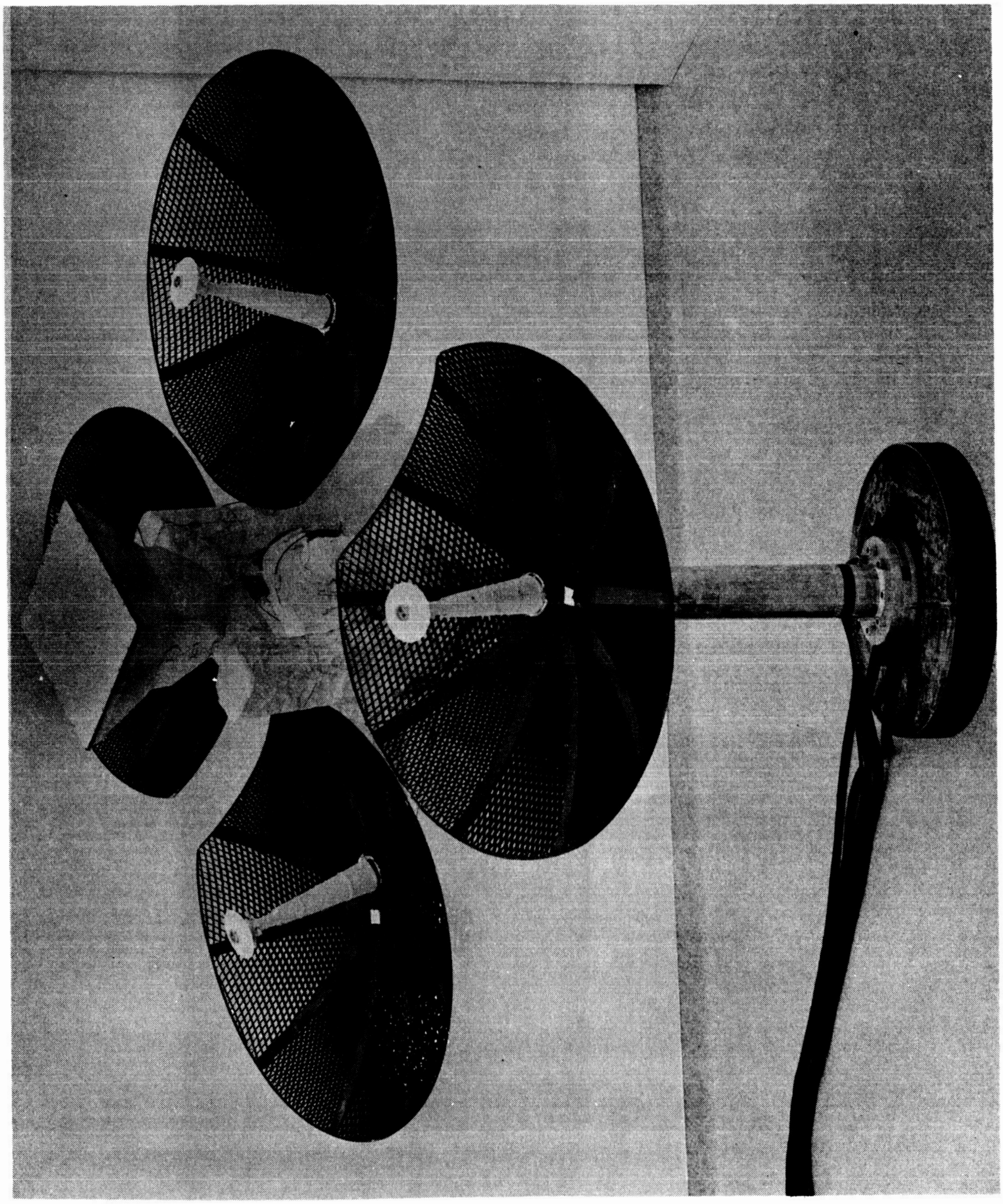


Figure 5.12. Proposed Mission Module to Probe Antenna

5.3.2 Reliability Analysis

1. The MMP Antenna, using the Apollo concept for probe communications only, would require no change in the design to provide commensurate reliability or probability of mission success. This conclusion is based simply on the duration of the proposed function. Indications are that the duration of probe communications is 400 hours. The reliability of the Apollo antenna configuration is based on a duration of 336 hours. Therefore, the probability of mission success of the Apollo antenna for probe communications is satisfactory without maintenance or repair required. That is, the contribution to P_{ms} will exceed 0.999 and, since this is a Criticality III function, this more than meets the objective.

2. The MME Antenna Reliability must be based on a full 16,550 hours of operation. It is a Criticality II system and must therefore achieve a higher probability of no uncompensable failures (P_s). The Apollo failure rates for the antenna functions were used and result in the reliabilities given in Figure 5.12/13 The resultant reliability for the baseline mission would be

$$P_s = P_a P_e P_{cu}$$

$$= 0.81$$

This is obviously too low and must be improved considerably. From the breakdown presented in logic of Figure 5.13, it is evident that these are three weak links limiting the reliability of the system.

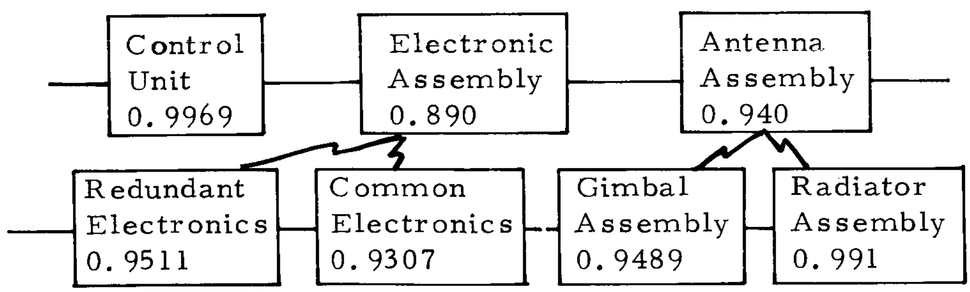


Figure 5.13. Reliability Logic, Antenna System

5.3.3 Availability Requirements Analysis

The MME antenna will require some sort of support to improve the P_s contribution to an acceptable level. Each of the weak links previously identified must be improved to equalize risk at about 0.998, which will assure a P_s of about 0.99. Each weak link is analyzed in the following.

1. The control unit of the antenna system provides a visual indication of the antenna pointing direction in spacecraft coordinates and the manual control of that pointing direction. Although the narrow beam width of the 19-ft antenna precludes the manual control of the antenna during the major portion of the mission, there are some instances where this capacity is desirable. Accordingly, this assembly was considered as a required part of the system.

   The critical components of the control unit are shown in Table 5.10. They consist of four resolvers and the terminal solder connections. Two of the resolvers are size 15, and two are size 11. The Apollo II failure rate is

$$\lambda_{c_u} = 0.0188 \times 10^{-6}, \text{ or } R = 0.99685$$

   for the mission. Since the solder connections are to terminals and the capability of repair is assumed, the failure rate of these connections are considered negligible in the 700-day mission. It can be seen that, even without the failure of solder connections, the unit failure rate still exceeds the apportionment. Consequently, the sparing considered was one each size 15 and one each size 11 resolvers. The result of this sparing is that P_s of the equivalent unit becomes $P_s' > 0.99999$.

   This is far above the requirement but sparing only one of the resolvers would be risky since both exhibit the same failure hazard and they are not interchangeable.

2. The Electronics Assembly contains the logic circuitry for the directing and automatic control of the antenna. Because of the previously mentioned narrow beam width of the antennae, this automatic function becomes more critical with mission extension attempts. The present construction of this unit is basically, 2 by 2 P/C logic modules mounted on four mother-boards. Figure 5.14 shows that the circuits are divided into two major groups—common and redundant. The common circuitry modules and/or components are listed in Table 5.11. The redundant circuits modules are listed in Table 5.12 and in logic form in Figure 5.15. From these it is evident that the baseline configuration contains some internal redundancy in standby operation. If a component fails in the primary circuit, relays allow the switching to the complete secondary circuit. This was required since no

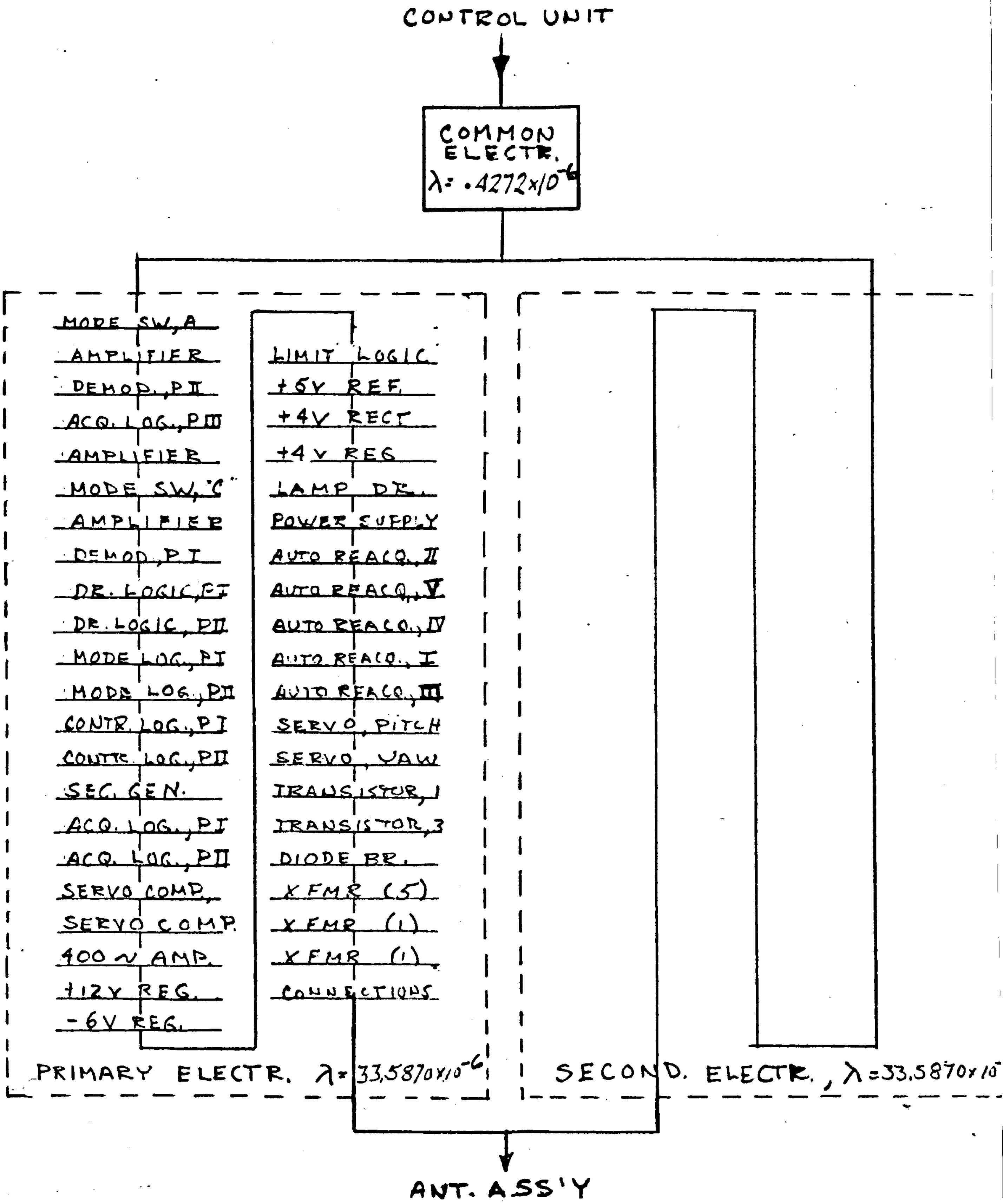


Figure 5.14. Electronic Assembly Logic Diagram

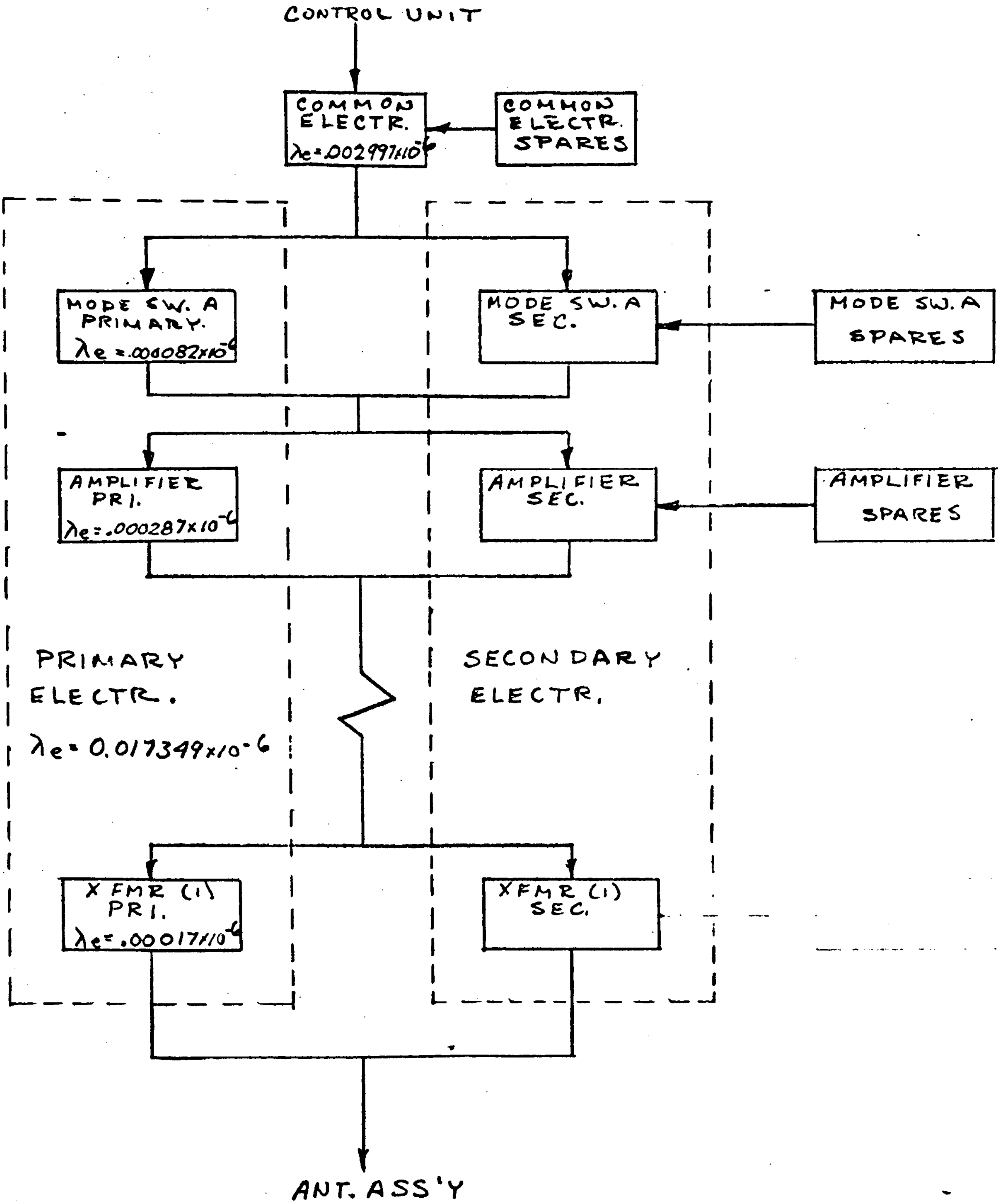


Figure 5.15. Exterior Electronic Assembly Logic Diagram

repairs were allowed for the lunar mission. If a different component fails in the secondary circuit (the nonredundant circuits), both systems have failed, and the antenna is inoperable with respect to automatic tracking. Adhering to this mode of operation (block redundancy), the probability of mission success is:

$$P_s = 0.8008$$

which is too low. The reliability of the redundant circuit alone is:

$$P_r = 0.81444$$

where

$$\lambda_r = 33.5870$$

for each circuit. The reliability of the common circuit is

$$P_c = 0.993079$$

for

$$\lambda_c = 0.4272$$

By providing for the replacement of the electronics assembly <u>in toto</u> the P_s would be raised to:

$$P_s = P_c P_r (2 - P_r)$$

$$= 0.96344$$

As can be seen, the attainment of the required P_s would require the sparing of two or more of these assemblies, with the attendant structure, for the baseline mission. However, this method of sparing is only required when assuming zero repair capability or when the same components or subassembly in both the primary and secondary circuit fail; i.e., the failures are not random.

In the analysis of this assembly, the provision for maintenance and repair no longer requires the present block redundancy operation. Considering that a failure of a single part in a circuit does not

Table 5.10. Antenna Control Unit

Function Assembly	Duty Cycle (hours)	Apollo II Assembly $\lambda \times 10^{-6}$	Mars Mission Spared Equivalent	Spares
Resolver, pitch	16,500	0.004	0.000005	0 (2 out of 3 operating)
Resolver, yaw	16,500	0.004	0.000005	1
Resolver, size 11	16,500	0.004	0.000005	0 (2 out of 3 operating)
Resolver, size 15	16,500	0.004	0.000005	1
Connections	-	0.0028	-	-
Total	-	0.0188	0.000020	2

Table 5.11. Electronic Assembly, Common Circuit

Functional Assembly	Duty Cycle (M)	Apollo II $\lambda \times 10^{-6}$	Mars Mission Report (16,500 hr)	
			Spared Equivalent	Spared Required
Instrument servo, pitch	16,500	0.05130	0.000862	1
Instrument servo, yaw	16,500	0.05130	0.000862	1
Transistor Board No. 3	16,500	0.01980	0.000222	1
Resolver	16,500	0.0004	0.000400	-
Resolver	16,500	0.0004	0.000400	-
Inductor	16,500	0.0020		
Capacitors (8)	16,500	0.0400	0.000040	8
Transformers (1)	16,500	0.0500	0.000050	1
Transformers (1)	16,500	0.0500	0.000050	1
Connectors	16,500	0.0990	-	
Connections	16,500	0.0480	-	
Relays	16,500	0.0150	-	
		0.4272	0.002997	

Table 5. 12/13 Electronics Assembly, Redundant and Circuits

Functional Assembly	Duty Cycle (hr)	Apollo II $\lambda \times 10^{-6}$	Mars Mission Requirement (16, 500 hrs)	
			Spared Equivalent	Spares Required Weight (lb)
Mode Switch "A"	16,500	0.6672	0.000082	1
Amplifier	16,500	1.0145	0.000287	1
Demodulator P. II	16,500	0.4032	0.000018	1
Acquisition logic P. III	16,500	0.9743	0.000255	1
Amplifier	16,500	1.0168	0.000289	1
Demodulator, P. I	16,500	2.0754	0.002393	1
Drive logic, P. I	16,500	1.8669	0.001851	1
Drive logic, P. II	16,500	1.0680	0.000346	1
Mode logic, P. I	16,500	0.3970	0.000235	1
Mode logic, P. II	16,500	0.9479	0.000235	1
Central logic, P. I	16,500	0.5656	0.000050	1
Control logic, P. I	16,500	0.8218	0.000153	1
Secret generator	16,500	0.7349	0.000110	1
Acquisition logic, P. I	16,500	1.3982	0.000744	1
Acquisition logic, P. II	16,500	1.0885	0.000354	1
Servo component	16,500	1.1615	0.000429	1*
Servo component	16,500	1.1615	0.000429	1*
400 cycle amp	16,500	0.4769	0.000030	1
+12v req.	16,500	1.3965	0.000742	1
-6v req.	16,500	0.4639	0.000028	1
Limit logic	16,500	0.3328	0.000010	1
+5v reference	16,500	1.4377	0.000808	1
+4v rectified	16,500	0.4956	0.000034	1

Table 5. 12/13Electronics Assembly, Redundant and Circuits (Cont)

Functional Assembly	Duty Cycle (hr)	Apollo II $\lambda \times 10^{-6}$	Mars Mission Requirement (16, 500 hrs) Spared Equivalent	Spares Required Weight (lb)
+4v Req.	16,500	0.7578	0.000120	1
Lamp driver	16,500	0.3258	0.000010	1
Power supply, R. F.	16,500	0.2184	0.000003	1
Automatic Reacquisition, P. II	16,500	0.1174	0.000001	1
Automatic Reacquisition, P. V	16,500	0.3260	0.000010	1
Automatic Reacquisition, P. IV	16,500	0.5394	0.000044	1
Automatic Reacquisition, P. I	16,500	0.2978	0.000007	1
Automatic Reacquisition, P. III	16,500	0.5363	0.000043	1
Instrument servo, pitch	16,500	1.8969	0.001837	1*
Instrument servo, yaw	16,500	1.8969	0.001837	1*
Transistor board No. 1	16,500	2.2835	0.003171	1
Transistor board No. 3	16,500	0.3120	0.000008	1
Diode bridge	16,500	0.1984	0.000003	1
Transformers (5)	16,500	0.2500	0.000010	1
Transformers (1)	16,500	0.0500	0.000170	1*
Transformers (1)	16,500	0.0500	0.000170	1*
Connections	16,500	0.1200		
TOTAL	33, 5870	33.5870	0.017349	42 39*

*Common elements which, because of interchangeability, permit reducing spares required by one.

nullify the usefullness of balance of the components, the logic diagram becomes redundant by part or by module. Figure 5.13 shows the effect of considering the redundant portion of the assembly as being redundant by module. The effective failure rate indicated is then dependent on only one out of three modules operating when sparing is included and based on operational redundancy. The suggested mode of operation permits the failed section to be repaired with the spare unit while the redundant circuit is operating. This procedure continues until the spares are exhausted. The assumption is that one of the two circuits then becomes a source of spares for the other circuit. Considering the above sparing and module redundancy, the contribution to P_s is :

$$P_s \geq 0.9997$$

This is in excess of the requirement but, because of the limited interchangeability of modules and their evenly distributed failure rate, each module must have a spare even to exceed 0.998. This required a total of 39 spare modules.

3. <u>The Antenna Assembly</u> consists of two major subassemblies—the gimbal assembly and the radiating assembly. The major critical components of these two assemblies are as shown in Table 5.14.

   The radiating assembly contains both the wide beam and narrow beam array and the strip line and switching networks. The failure rates of the component and/or subassemblies permit this function to exceed its requirement; the resulting reliability is over 0.999.

   The gimbal assembly contains the drive motors and position indicators of the three-axis system. Sparing, if done at all, must be as a total assembly replacement rather than a component replacement concept. The gimbal is a complex assembly (3-axis drive and angular indicator) designed for minimum weight (Figure 5.16). It is a structural key in the automatic tracking loop. The redesign of this unit to accommodate component replacement, primarily the motor-tach-brake sets of each axis could possibly result in an overall weight penalty rather than a savings because of the added structure required to accommodate accessibility. The assembly and disassembly are hazardous to the components; i.e., the constituent components are susceptible to assembly or installation failures.

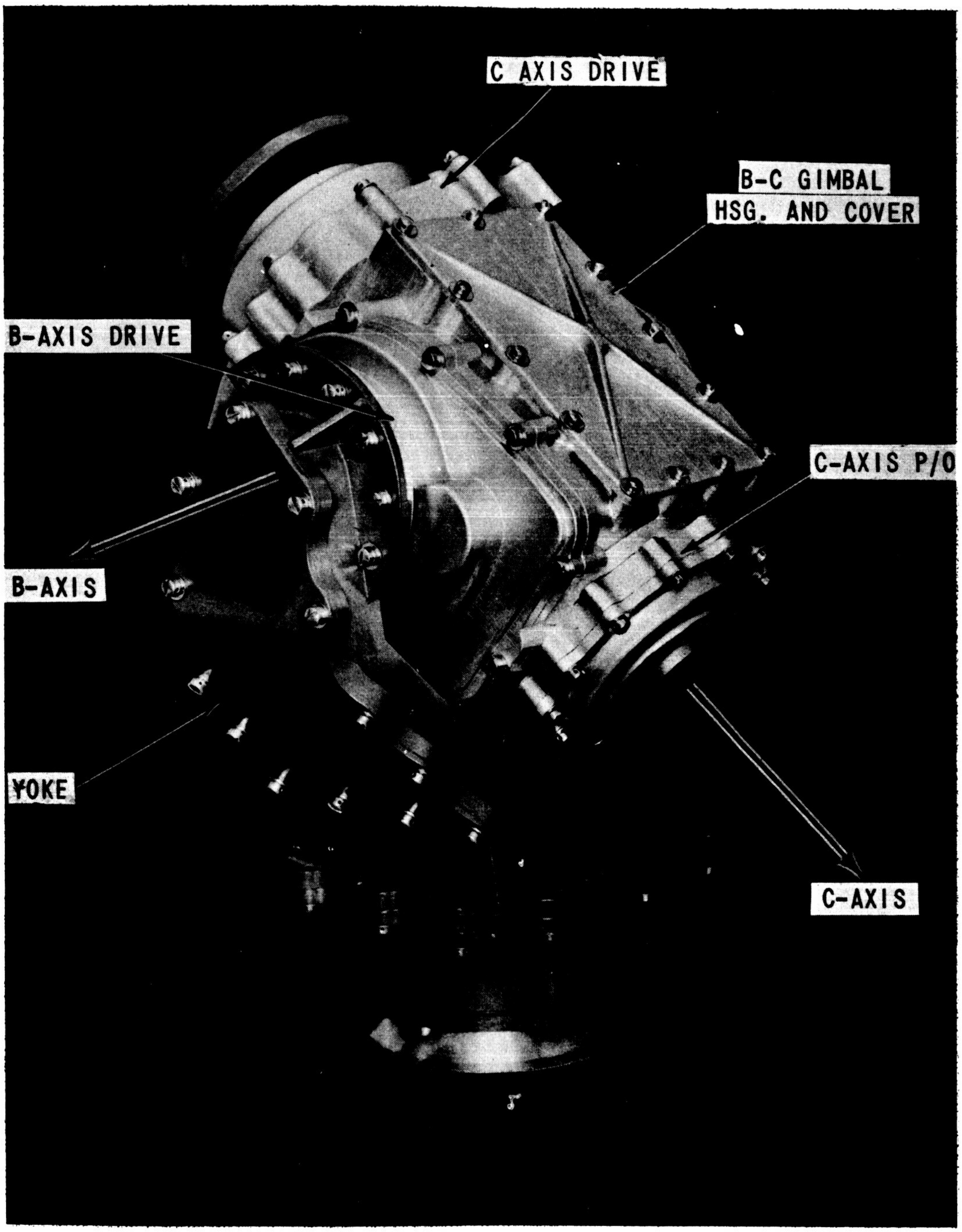


Figure 5.16. Initial Space Gimbal Mock-up (without insulation)

Table 5.14. Antenna Assembly (MME) Availability Requirements Analysis

Functional Assembly	Duty Cycle (M)	Apollo II Assembly $\lambda \times 10^{-6}$	Baseline Mission Reliability	Recommended Supports
R. F. Components				
Feedhorn assembly (4)	16,500	0.009632		
Hybrid (4)	16,500	0.009200		
W.B. comparator	16,500	0.006034		No spares required
N.B. comparator	16,500	0.004000		
Dummy load (4)	16,500	0.004000		
Transfer switch	16,500	0.002020		
Dual diplexer	16,500	0.001037		
Miscellaneous	16,500	0.00060		
Subtotal		0.0054040	0.99991	
Gimbal Components				*One spare gimbal assembly
A-axis	16,500	0.15890		
B&C axis	16,500	0.153185		
Subtotal		0.312085	0.99485	0.999999
Total		0.3174	0.99477	0.99991

*Not necessarily required or recommended. Spare antenna would have the same effect.

These data indicate that the antenna assembly would be marginal in its capability to meet the allowable contribution to P_S; however, since the other functions were much higher than required, the resulting antenna system P_S contribution will still only be about 0.994. The other alternative is to spare the total antenna assembly, which increases the P_s to 0.9997. The electronics assembly again becomes the weaker link. This is not recommended because of the inherent redundancy achieved by using the MMP antenna as a backup assembly; this effectively meets the system requirements.

5.3.4 MME Antenna Maintainability Inferences

The recommended maintenance concept requires replacement of the electronic assembly modules and the control unit resolvers. Both of these are expected to be easily accessible from inside the spacecraft. The resulting maintenance concepts infer that only a minimum of redesigning and testing is required to adapt the Apollo CSM antenna system to fulfill the planetary mission requirements without sacrificing the existing system reliability. The primary design change that would be required of the system would be in the electronics assembly. Presently, the zero maintenance criterion has resulted in a potted assembly. To accommodate the repairs and maintenance referred to herein, it would be required that the method of potting be revised to allow the rapid or simple removal and replacement of the 2 by 2 logic module from the motherboards: one of the three motherboards is shown in Figure 5. 19.

In addition, some 2 by 2 module configurations would require changing to accommodate the repair of one circuit without disrupting the operating circuit if this concept were adopted.

5.3.5 Conclusions and Recommendations

The Apollo antenna will meet the requirements of the mission module to probe and earth entry module to earth functions without significant change and without provisions for maintenance. The same antenna system with modification will also meet the MME system requirements. Modifications required include the replacement of the feed horn and dish assembly with a single unfurlable 19-foot dish. This has been built by Dalmo Victor Company and tested by the General Electric Company.

The application of the Apollo II CSM antenna system for primary earth communications, either as backup or concurrent with the 19-foot antenna, requires that it be maintained and/or repaired to meet with the suggested objectives.

The MME antenna system alone is expected to achieve a P_s of at least 0. 994 with the recommended spares complement. However, to augment this capability, the MMP antenna can be used instead of the MME in the event of failure, but at a slower data rate.

The result of this premise is reflected in Figure 5. 17, and the actual probability of mission success becomes

$$P_s = 0.9999998$$

when considered as block redundant and,

$$P_s = 0.99999999$$

when optimum commonality between the two antennas is accommodated, as shown in Figure 5. 18. This means that there is less than one chance in 10^6 of not having a usable antenna during the baseline mission.

Table 5. 15. MME Antenna Spares Requirements

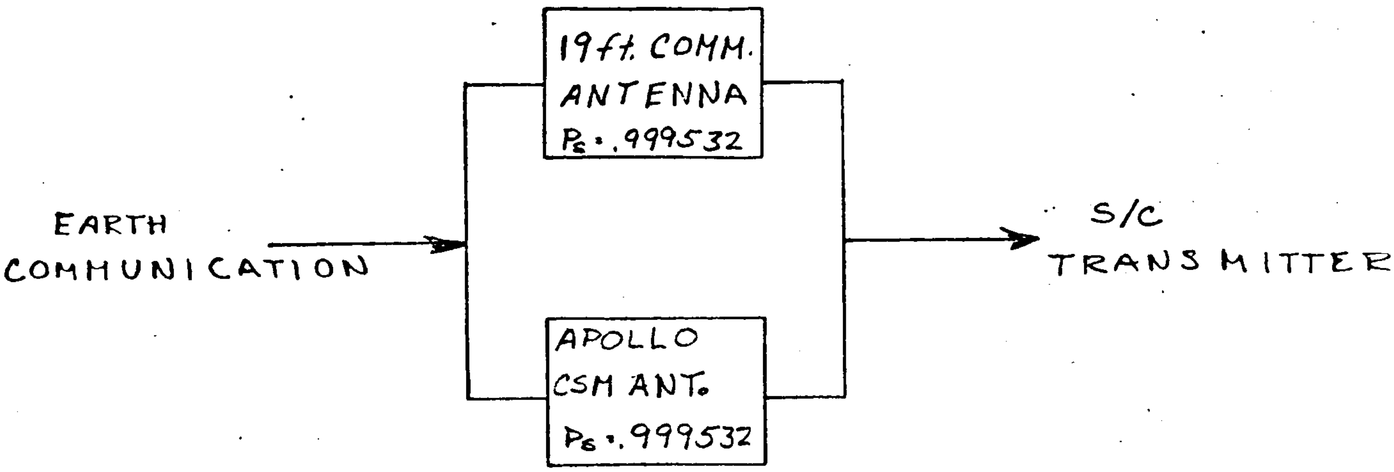
Components	No. of Spares	Weight (lb)
Resolvers	2	0.5
Electronic modules*	39	2.5
Totals	41	3.0
*See previous tables for breakdown		

5. 4 THE EXTENTION—RETRACTION SYSTEM

Note: These data used herein were derived from SD studies, References 1. 1 and 4. 3.

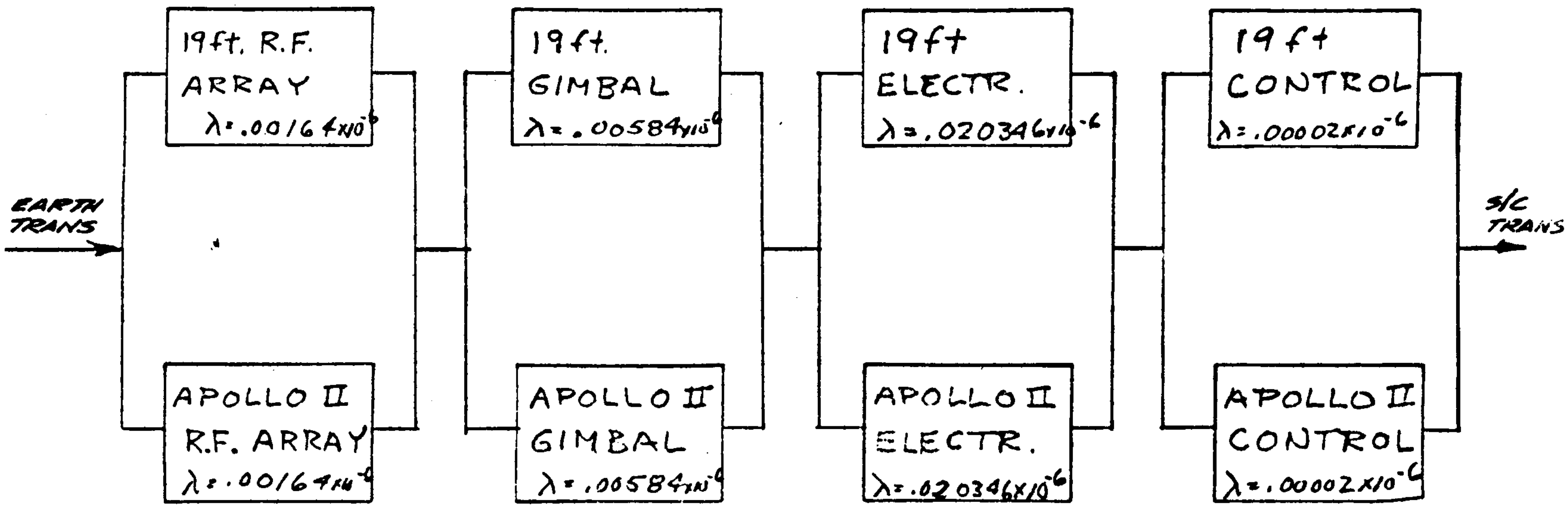
5. 4. 1 Functional Requirements

The operational concept is depicted by Figure 5. 20 the baseline mission spacecraft. Mechanical systems are provided to permit separation and extension of the earth entry module (EEM) from the mission module for the spin-up mode (extended position). Spin-down is accomplished by a system of motor dirven cable drums that reel the module from the extended position to the unextended position.



$$P' = P_s(2 - P_s)$$
$$= .9999998$$

Figure 5.17. Primary Communication Antenna Logic Diagram



$$P_S = P_A \cdot P_G \cdot P_E \cdot P_C$$

$$P_S = .999\,999\,9$$

Figure 5.18. Primary Communication Antenna Opt. Rel. Logic

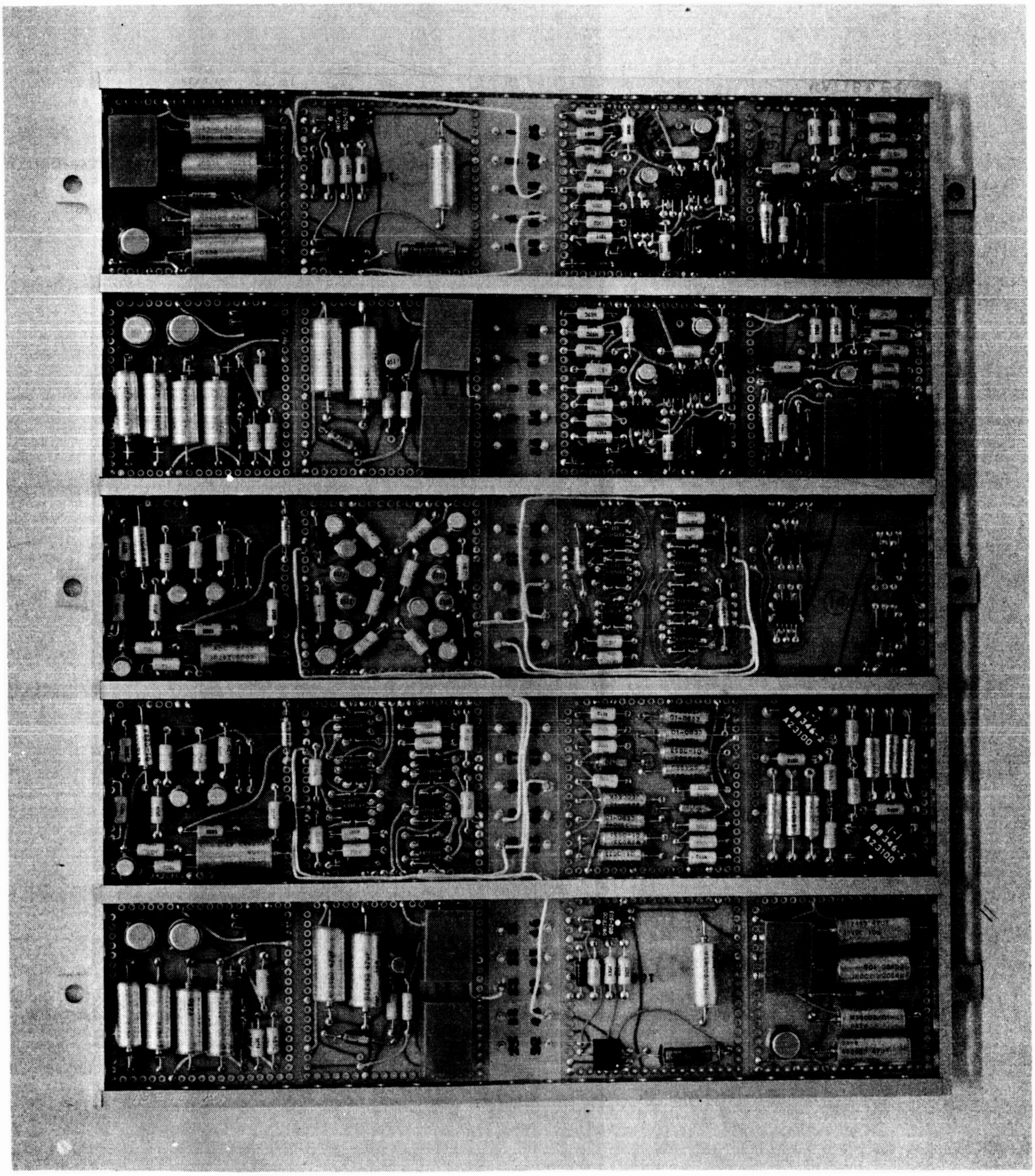


Figure 5.19. Detail of Mother Boards, Antenna Control Function

Other concepts for separation and extension of the modules, including telescoping tubes, were considered. However, the cable system, which appears feasible from the dynamic aspect and is considerably lighter than the other concepts, was selected as the baseline concept.

As the modules are separated, spin-up is started and progressively increased until maximum separation and proper cable tension are achieved. Timed thruster firing is used to place the module in the spin-up mode and establish the required artificial gravity condition. Despin will be accomplished progressively in the reverse order. As a backup or alternative method to provide an artificial gravity condition, counter-rotation centrifuges are installed in the mission module.

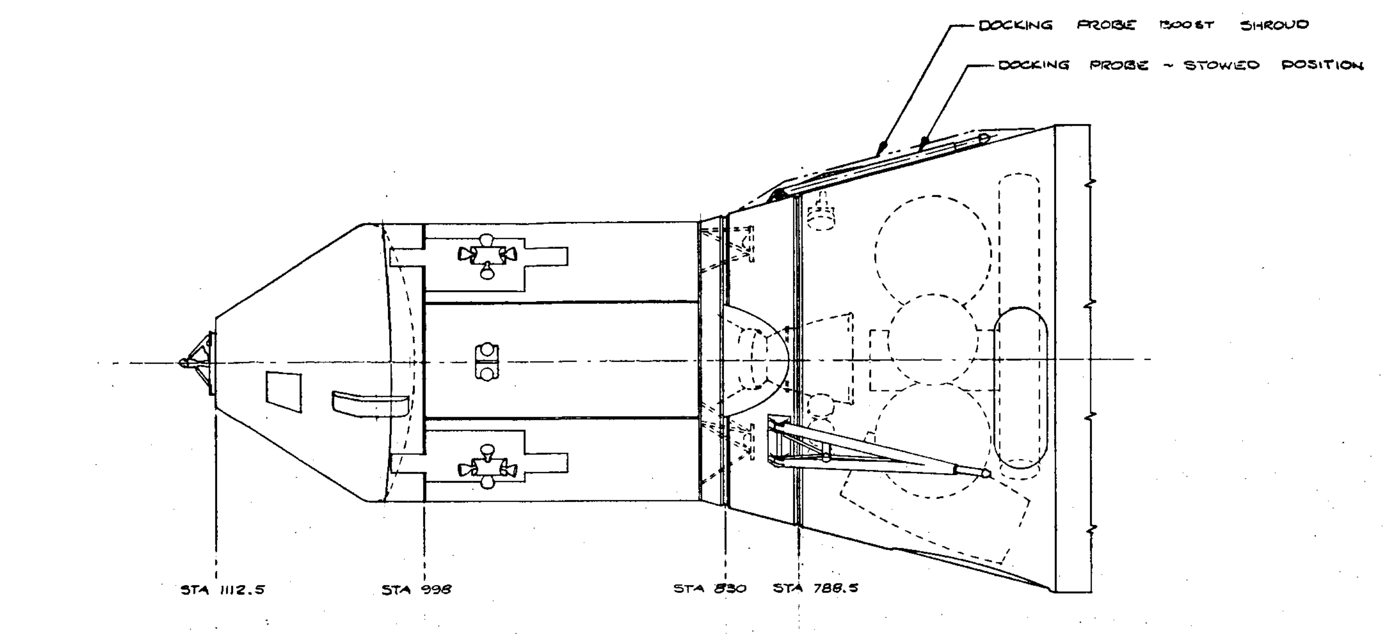
The cable assembly, reflected in the reliability block diagram (Figure 5.20), consists of three retraction cables and six stabilization cables. Each cable is connected to a cable drum equipped with its own driving power, gear box, clutch, and control. After separation has been effected, the stabilization cables are placed in tension and their drive units remain dormant for the remainder of the spin period. However, each cable is equipped with a dash pot to dampen and absorb vibrations due to crew movement, etc. Redundant electric drive motors have been assumed because of the uncertainties associated with extended space operation of exposed rotating components. Although both ac and dc cricuits are available ac motors provide a weight advantage and were selected for purposes of the analysis.

To limit the coning motion (wobble) associated with cable-linked space craft, a viscous fluid damper is employed. This system makes use of the on-board water and multiple small tubes to achieve the required damper effect.

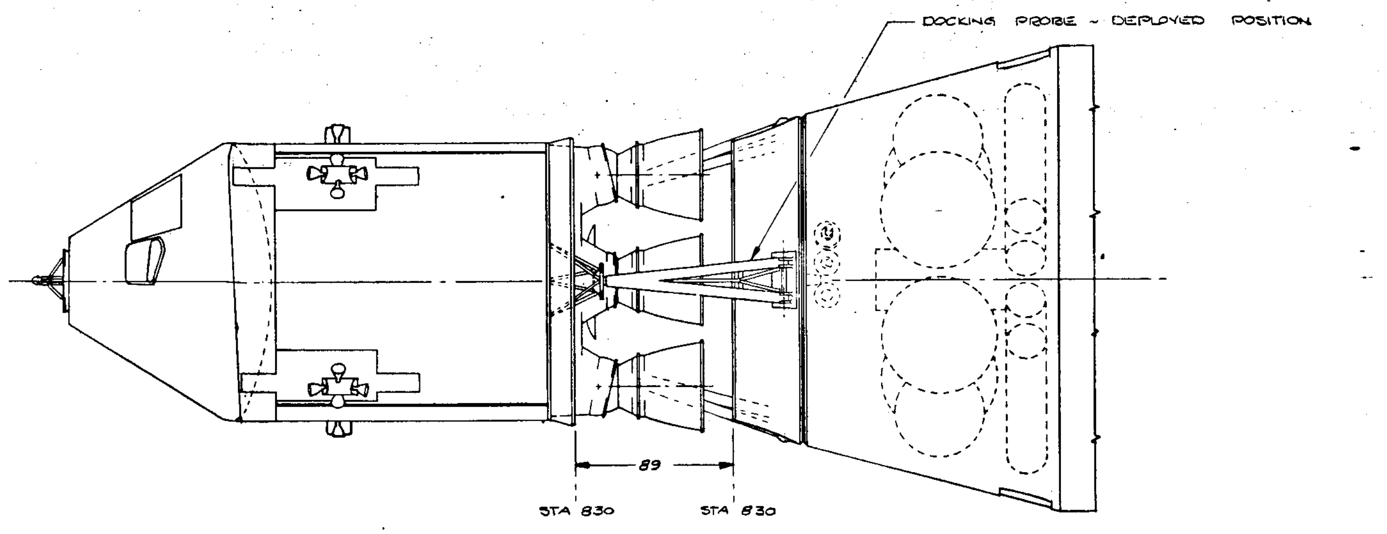
Following injection from earth orbit to planet trajectory, the command module will be extended and placed in the spin mode. The number of spin-up and spin-downs will depend upon the number of times required for navigation data acquisition and ΔV course correction. It has been assumed from the established mission events (Section 2) that the spin-up and spin-down modes will occur at least four times each on the outward and return trips of the mission plus a test spin-up and spin-down prior to Martian injection. Approximately 16,000 hours will be spent in the spin-up mode.

5.4.2 Reliability Analysis

The basic structural subsystem, including the separation and retraction mechanism (Figure 5.20) has a high inherent reliability, although certain design details remain to be resolved. Table 5.16 lists the components and associated operating times of the cable assembly. The operating time is



EARTH BOOST CONFIGURATION



INJECTION CONFIGURATION

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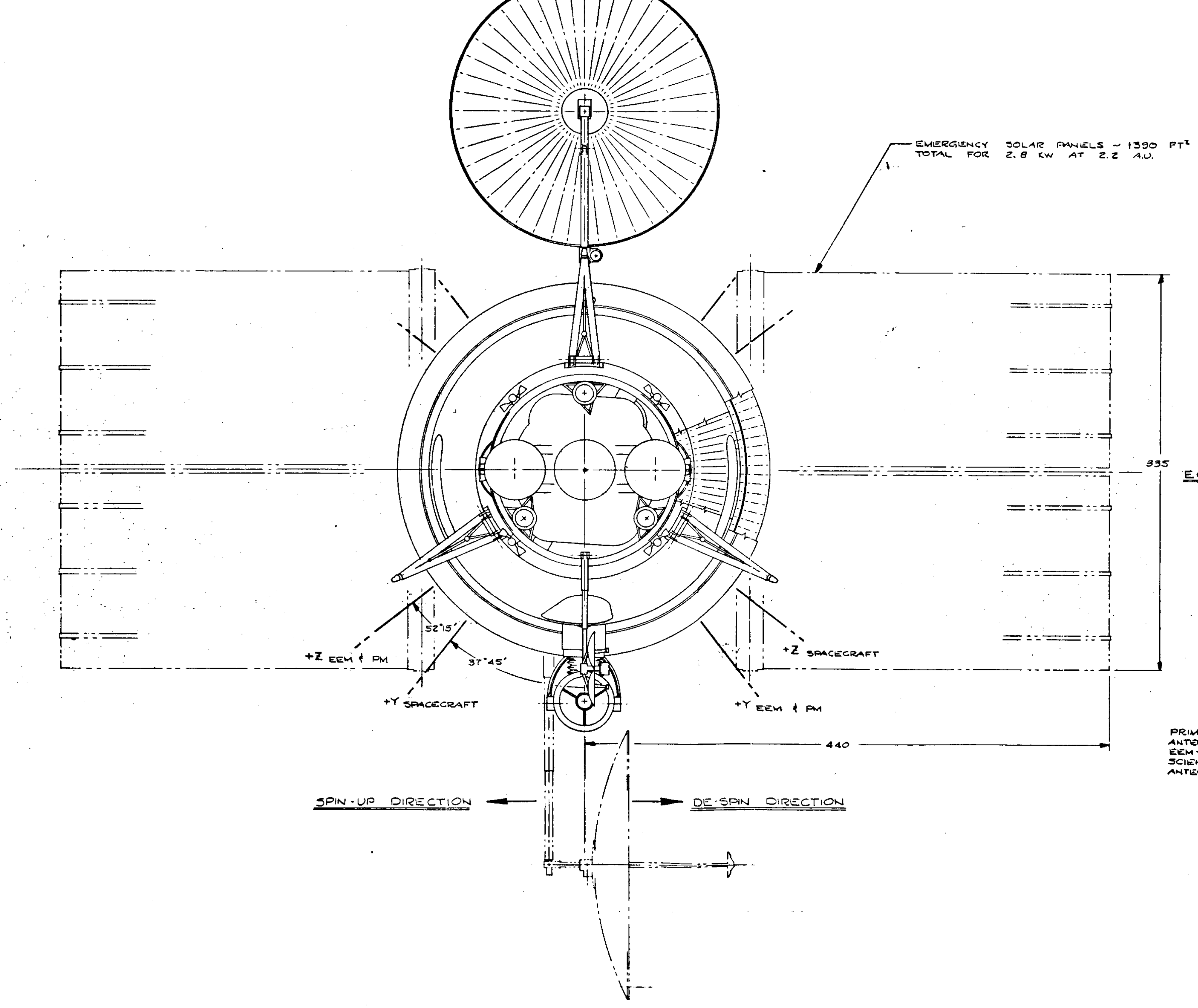

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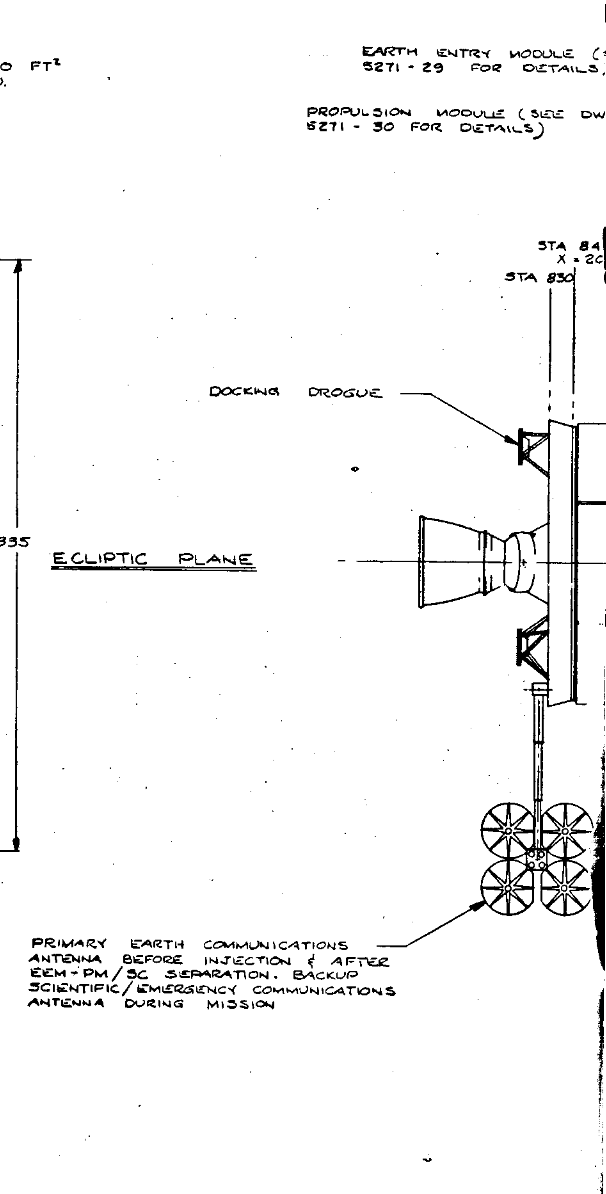

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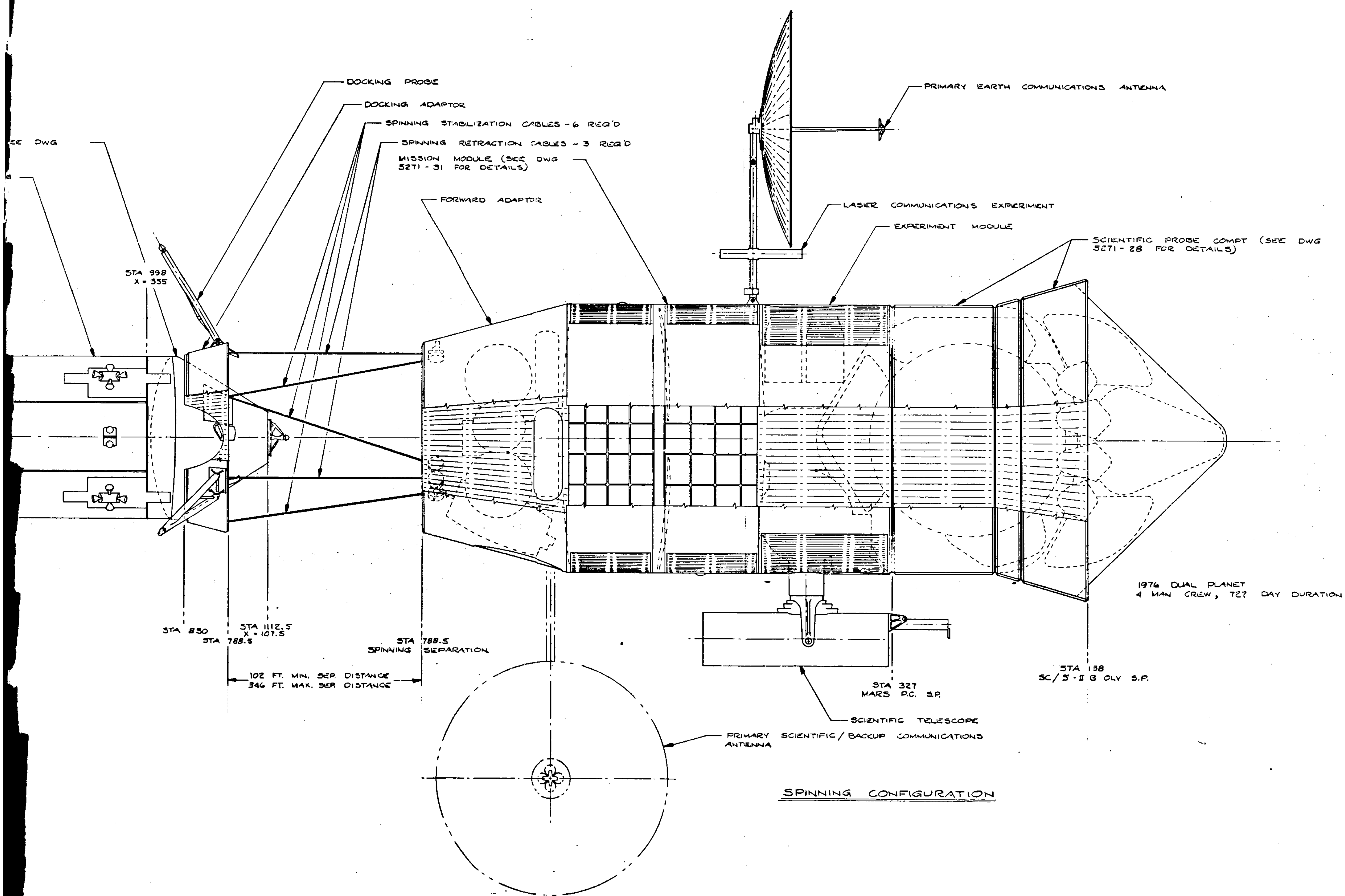

Figure 5.20. Vehicle Configuration

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based on 16,043 hours in the spinning mode with up to 17 spin-despin cycles of 1.2 hour each (worse case). There are considerable redundancies in the mechanism. For example, stabilization cables can act in lieu of a failed retraction cable. Because of the many possible combinations and the current design status, it has been assumed for simplicity that the stabilization cables can act as backup to the retraction cables. Based on these assumptions, the reliability of the overall is estimated to be 0.9998.

Table 5.16. Components and Associated Operating Times of Cable Assembly

Item	Number of Units	Operating Time, Cycles
Motor, control, and clutch	18	20.4
Gear boxes	9	20.4
Cables	9	16,043
Dash pot	9	1604*
Drums	9	16,043

*Assumes 10 percent total time

This will meet and exceed the P_s requirement, particularly when the functional redundancy of section 2.2 is considered. Maintenance and/or repair is not expected to be a requirement.

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VI. RECOMMENDATIONS AND CONCLUSIONS

6.1 SYSTEM DESCRIPTION

The baseline mission has been described in Volume I of this report in functional flow form. Further details are pertinent to the subsystems and are discussed in conjunction with the respective subsystem analysis of Section IV.

The baseline spacecraft system was taken from NAS8-18025 and expanded upon to the next level of detail. The mission spacecraft is as depicted in Figure 6.1. It is assumed to have the artificial gravity capability since this represents the most stringent system requirements. The mission safety/success goals were also derived in Volume I of this report, and the associated system level logic is given in Figure 6.2. The first ten systems (those in the heavy boxes) are required to assure safe return of the crew. The up data link is included because of its contribution to the guidance and navigation function as well as the central timing function. These systems are considered Criticality I systems (Table 6.1) because inability to perform any of these functions would endanger the crew.

Five of the systems contribute to the well being of the crew because, in most circumstances resulting from a failure in these functions, the crew could still return safely. These are included in the availability assessment and are considered Criticality II systems (Table 6.2). Artificial gravity systems are included herein because they are not essential to crew survival.

The scientific systems are Criticality III systems. These include all of the functions associated with the scientific data, either through direct acquisition or via probes. No analysis has been devoted to those systems in this study because of the lack of detail systems descriptions.

6.2 MISSION CAPABILITIES AND SUPPORT REQUIREMENTS

6.2.1 For Planetary Missions

The analysis herein demonstrates the ability of contemporary systems to meet the challenge of extended space missions. Man becomes an integral part of the mission systems and, through his ability to compensate for unprogrammed events, the probability of safe return for the baseline mission can be raised from five chances in one hundred ($P_s = 0.05$) to better than ninety-nine chances in one hundred (0.993). As Table 6.1 indicates, this can be accomplished with less than 500 pounds of well chosen spares.

One of the most significant findings of the study is the identification and verification of a concept which facilitates selection of the correct spares and the associated maintenance and repair concept. Through application of the availability concept, the guess work is taken out of spares provisioning and test point selection; this is demonstrated by the comprehensive nature of the data in the tabular results presented under the respective subsystems of Section V.

The results of the study are summarized in Tables 6.1 and 6.2. These support requirements were identified by criticality class where:

Criticality I = Minimum functions required for crew safe return.

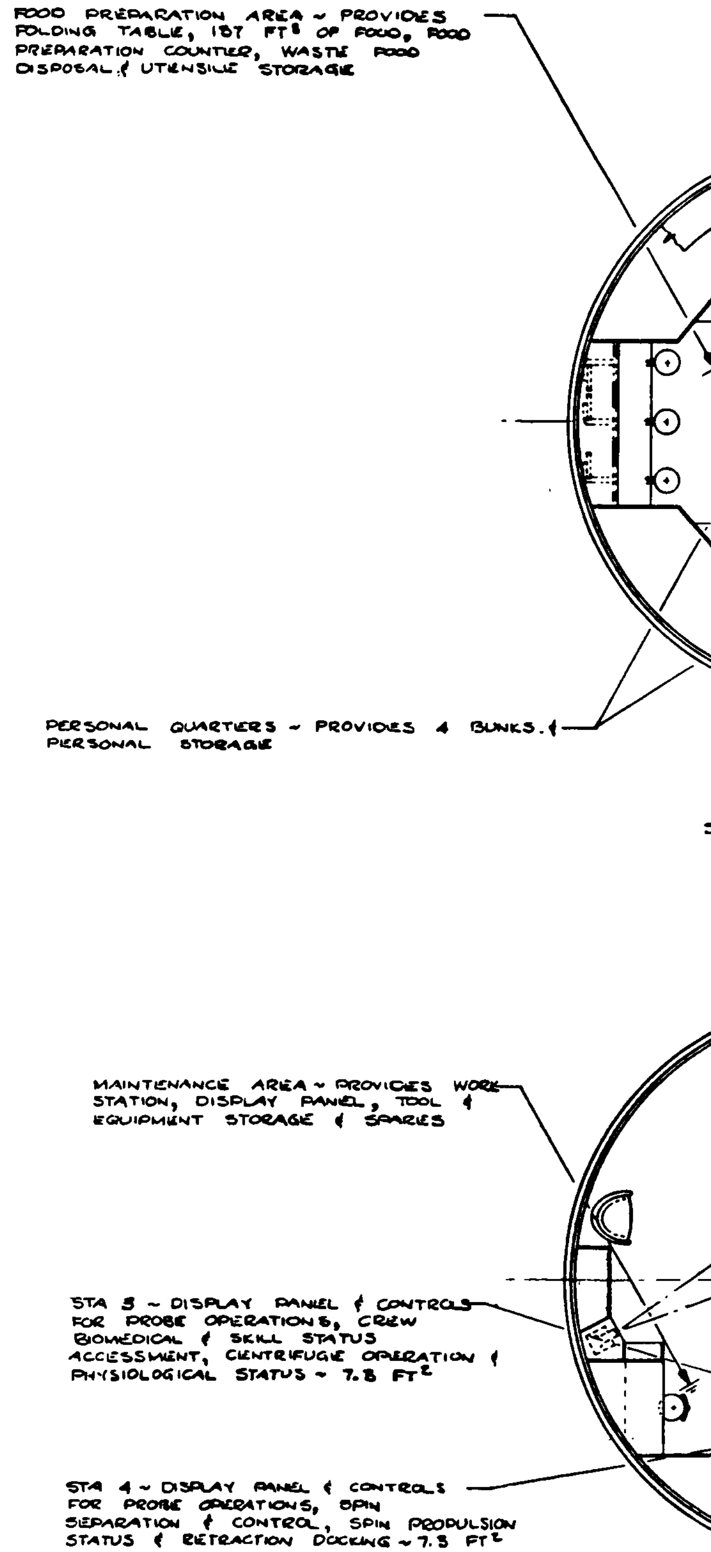
Criticality II = Functions required to perform the trip in the manner prescribed.

Criticality III = Functions required to perform all the mission objectives (not considered in this study).

The specific spares required may be found tabulated with the referenced report section. It should be realized that this analysis is based on a real, in most cases qualified, hardware design that will meet the mission requirements. Because it was possible to be specific in terms of the design and its reliability and because of the demonstrated veracity of much of the data, the conclusions drawn herein may be considered creditable. These same factors account for the seeming disparigence in the support requirements delineated herein and those estimated elsewhere.

These data indicate that only 167 maintenance and repair actions requiring 438 pounds of spares will bring the crew safely home with a probability of better than 0.993. Further, only 258 spares weighing less than 8 87 pounds will facilitate the conduct of the trip in the manner prescribed with a probability of better than 0.966. This means that out of 1000 baseline missions, only 36 would require a compromise in the operational concept and less than seven would encounter a problem that <u>may</u> result in loss of crew.

Not all of the potential weak links could be improved through a maintenance or repair action. From Table 6.1, items 4.5, 4.6, and 4.9 were improved (as noted in Table 6.3) by some form of design redundancy. Not



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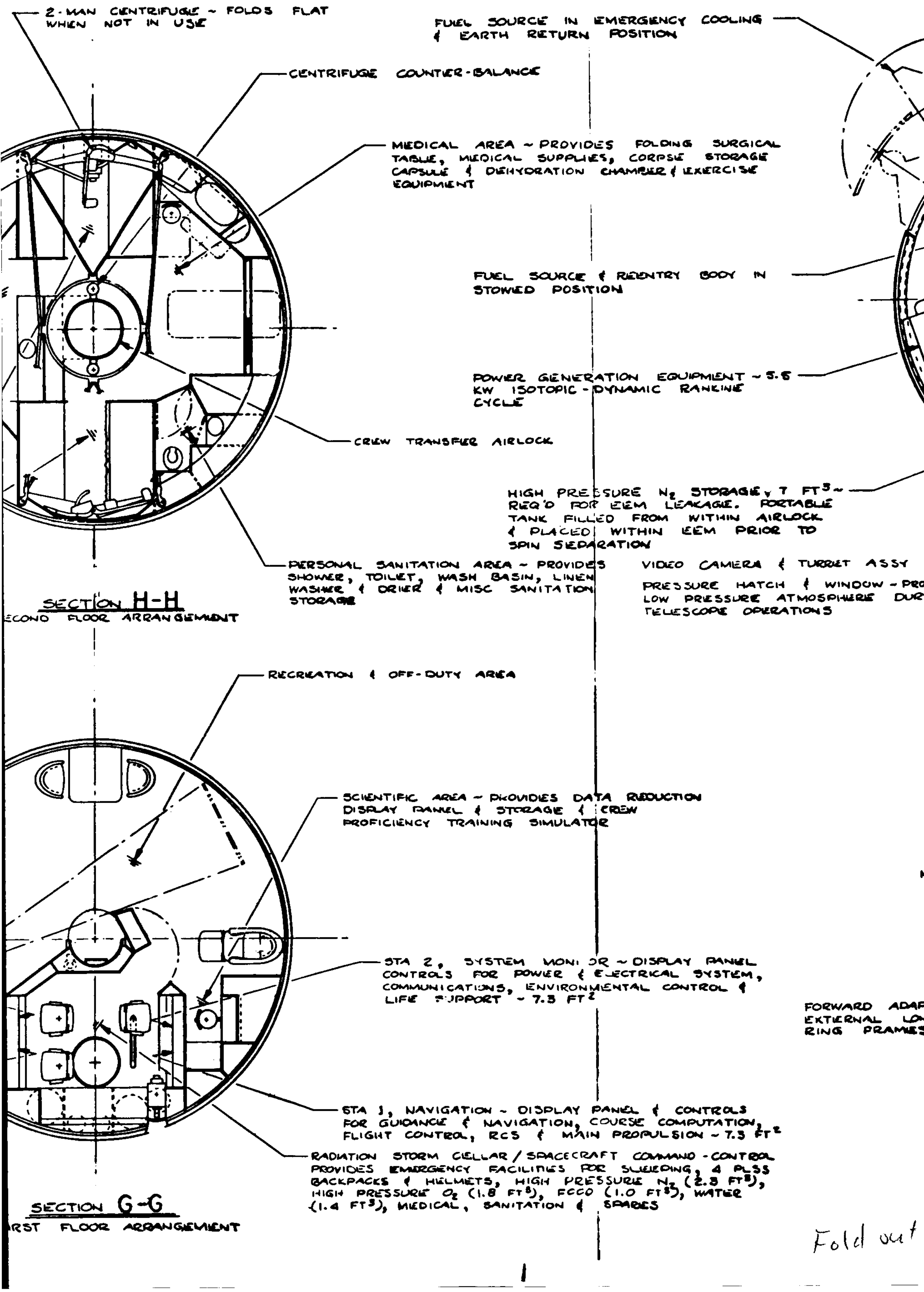
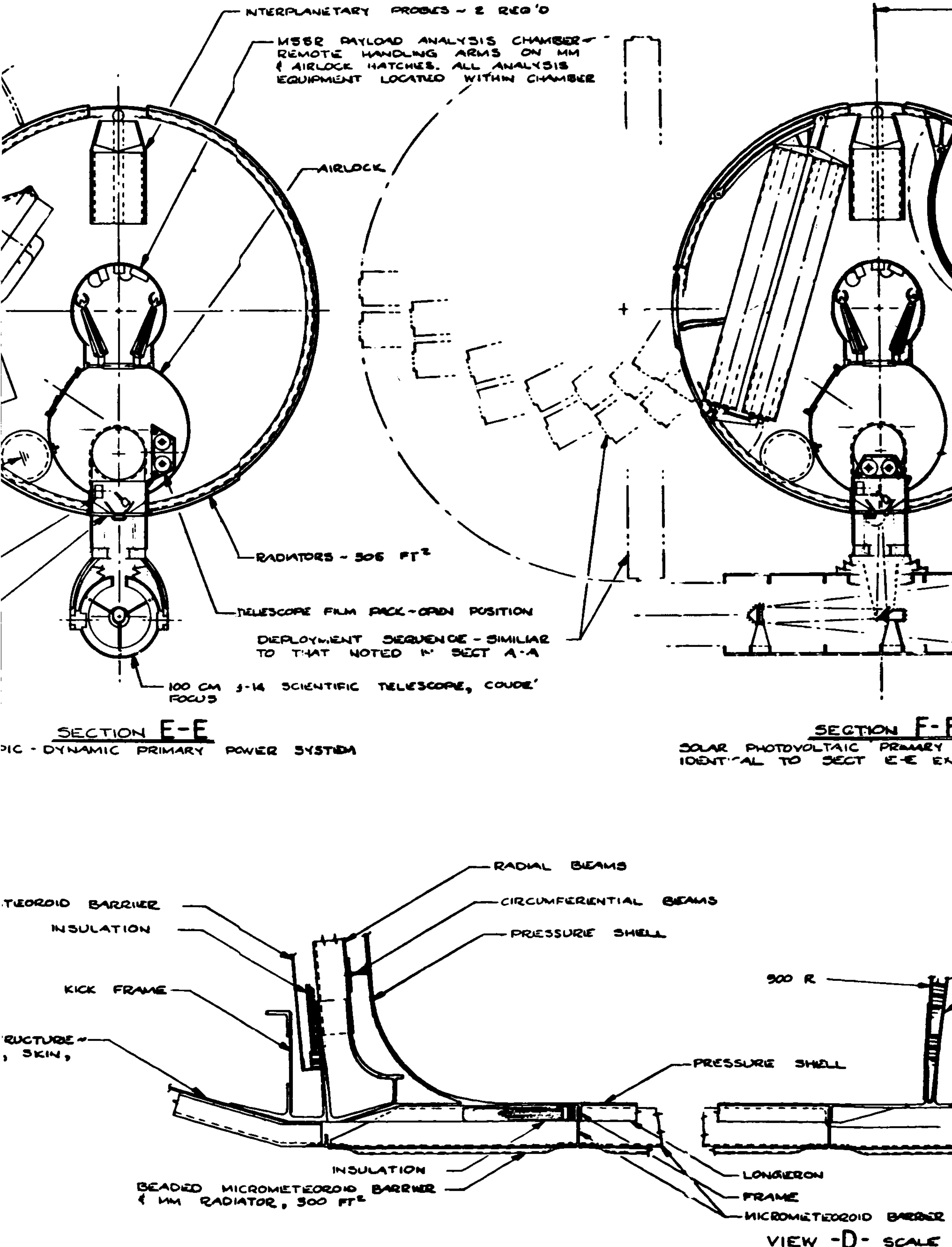

2-MAN CENTRIFUGE ~ FOLDS FLAT WHEN NOT IN USE
CENTRIFUGE COUNTER-BALANCE
MEDICAL AREA ~ PROVIDES FOLDING SURGICAL TABLE, MEDICAL SUPPLIES, CORPSE STORAGE CAPSULE & DEHYDRATION CHAMBER & EXERCISE EQUIPMENT
FUEL SOURCE IN EMERGENCY COOLING & EARTH RETURN POSITION
FUEL SOURCE & REENTRY BODY IN STOWED POSITION
POWER GENERATION EQUIPMENT ~ 5.5 KW ISOTOPIC-DYNAMIC RANKINE CYCLE
CREW TRANSFER AIRLOCK
HIGH PRESSURE N2 STORAGE, 7 FT3 ~ REQ'D FOR EEM LEAKAGE. PORTABLE TANK FILLED FROM WITHIN AIRLOCK & PLACED WITHIN EEM PRIOR TO SPIN SEPARATION
PERSONAL SANITATION AREA ~ PROVIDES SHOWER, TOILET, WASH BASIN, LINEN WASHER & DRIER & MISC SANITATION STORAGE
VIDEO CAMERA & TURRET ASSY
PRESSURE HATCH & WINDOW - PRO
LOW PRESSURE ATMOSPHERE DUR
TELESCOPE OPERATIONS
SECTION H-H
ECOND FLOOR ARRANGEMENT
RECREATION & OFF-DUTY AREA
SCIENTIFIC AREA ~ PROVIDES DATA REDUCTION DISPLAY PANEL & STORAGE & CREW PROFICIENCY TRAINING SIMULATOR
STA 2, SYSTEM MONITOR ~ DISPLAY PANEL CONTROLS FOR POWER & ELECTRICAL SYSTEM, COMMUNICATIONS, ENVIRONMENTAL CONTROL & LIFE SUPPORT ~ 7.5 FT2
FORWARD ADAP
EXTERNAL LON
RING FRAMES
STA 1, NAVIGATION ~ DISPLAY PANEL & CONTROLS FOR GUIDANCE & NAVIGATION, COURSE COMPUTATION, FLIGHT CONTROL, RCS & MAIN PROPULSION ~ 7.5 FT2
RADIATION STORM CELLAR / SPACECRAFT COMMAND-CONTROL PROVIDES EMERGENCY FACILITIES FOR SLEEPING, 4 PLSS BACKPACKS & HELMETS, HIGH PRESSURE N2 (2.3 FT3), HIGH PRESSURE O2 (1.8 FT3), ECCO (1.0 FT3), WATER (1.4 FT3), MEDICAL, SANITATION & SPARES
SECTION G-G
RST FLOOR ARRANGEMENT
Fold out


INTERPLANETARY PROBES ~ 2 REQ'D
MSSR PAYLOAD ANALYSIS CHAMBER - REMOTE HANDLING ARMS ON MM & AIRLOCK HATCHES. ALL ANALYSIS EQUIPMENT LOCATED WITHIN CHAMBER
AIRLOCK
RADIATORS ~ 505 FT²
TELESCOPE FILM PACK - OPEN POSITION
DEPLOYMENT SEQUENCE - SIMILIAR TO THAT NOTED IN SECT A-A
100 CM J-14 SCIENTIFIC TELESCOPE, COUDE' FOCUS
SECTION E-E
PIC - DYNAMIC PRIMARY POWER SYSTEM
SECTION F-F
SOLAR PHOTOVOLTAIC PRIMARY
IDENTICAL TO SECT E-E EX
RADIAL BEAMS
CIRCUMFERENTIAL BEAMS
PRESSURE SHELL
TEOROID BARRIER
INSULATION
KICK FRAME
RUCTURE ~
, SKIN,
300 R
PRESSURE SHELL
INSULATION
BEADED MICROMETEOROID BARRIER & MM RADIATOR, 500 FT²
LONGERON
FRAME
MICROMETEOROID BARRIER
VIEW -D- SCALE

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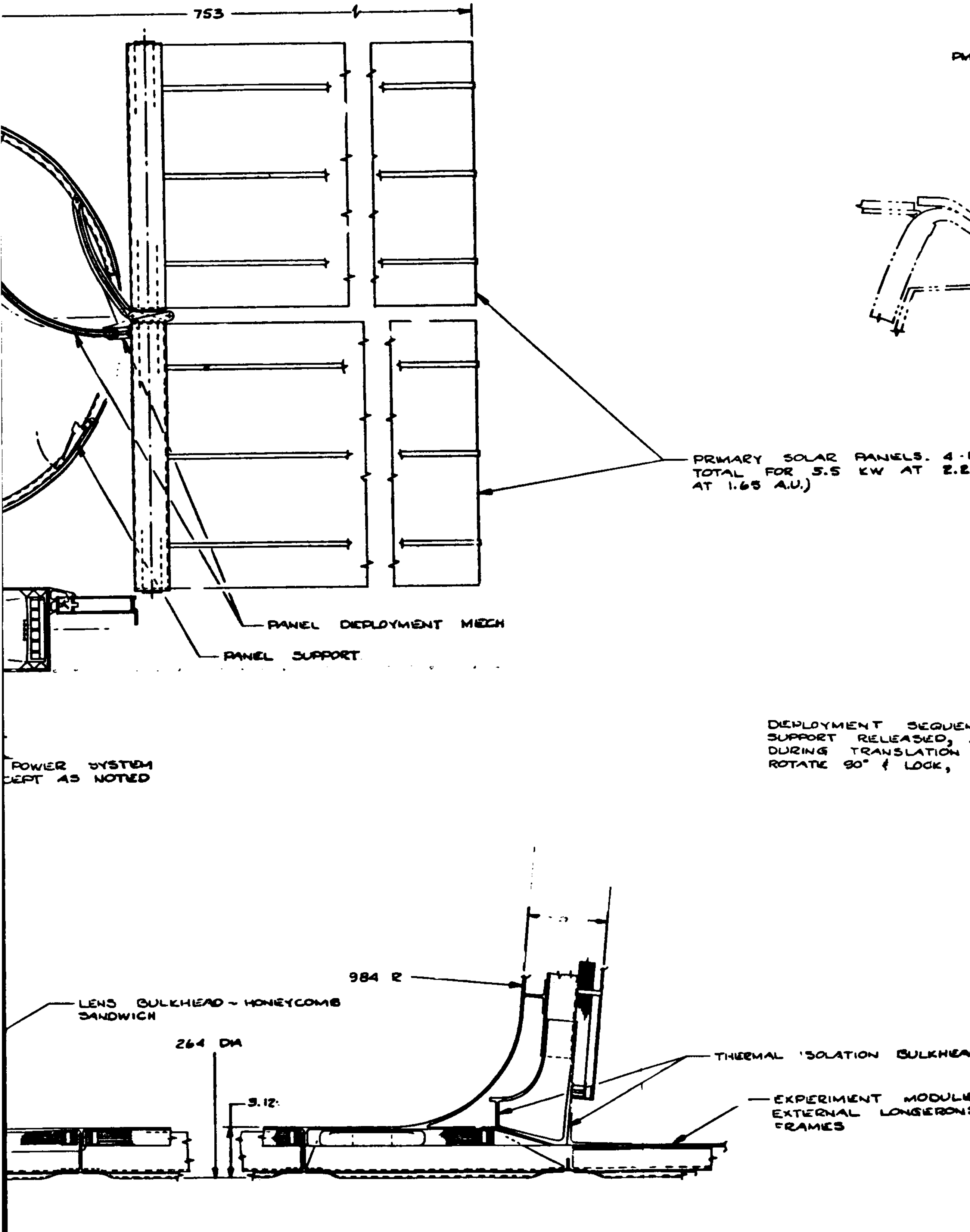

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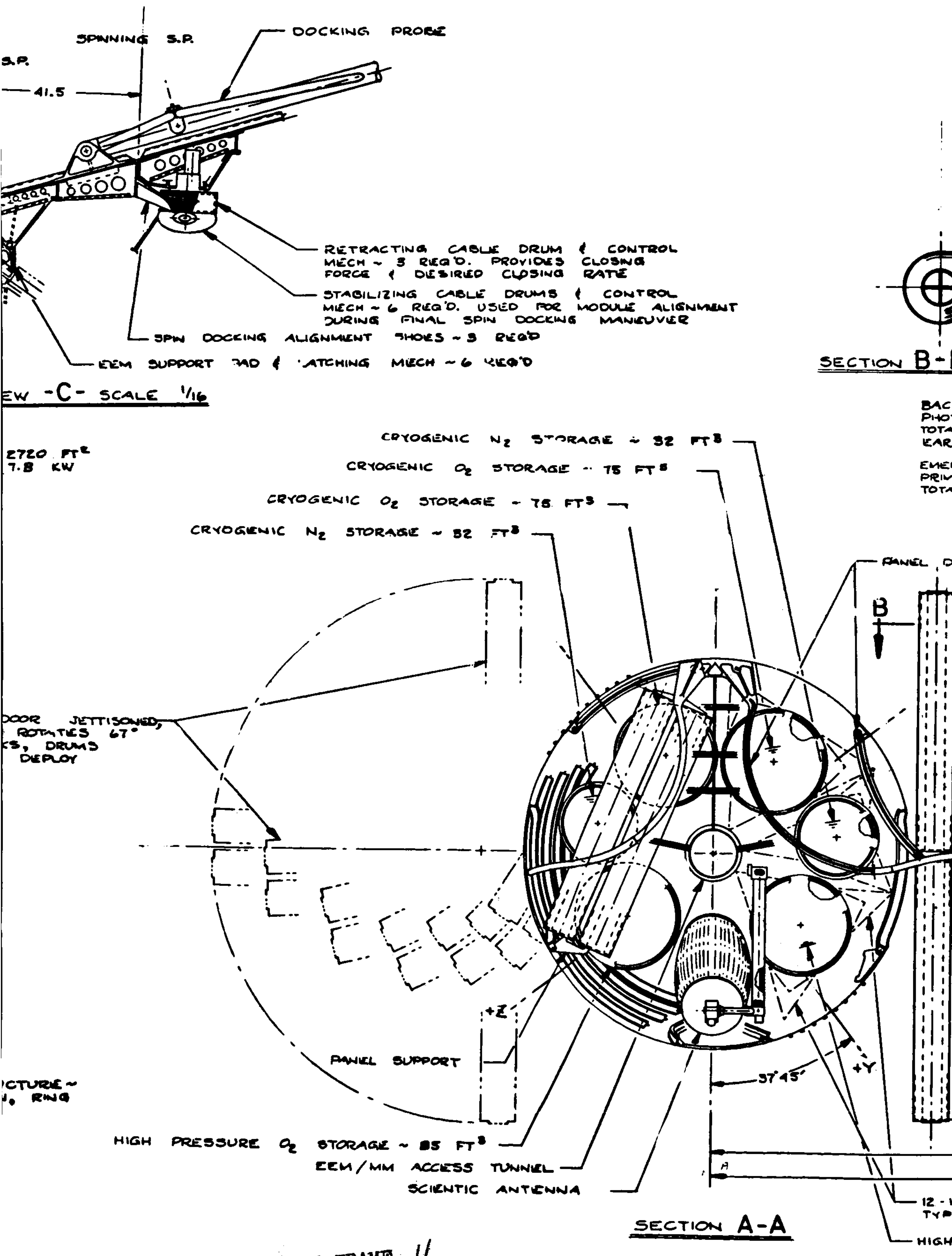

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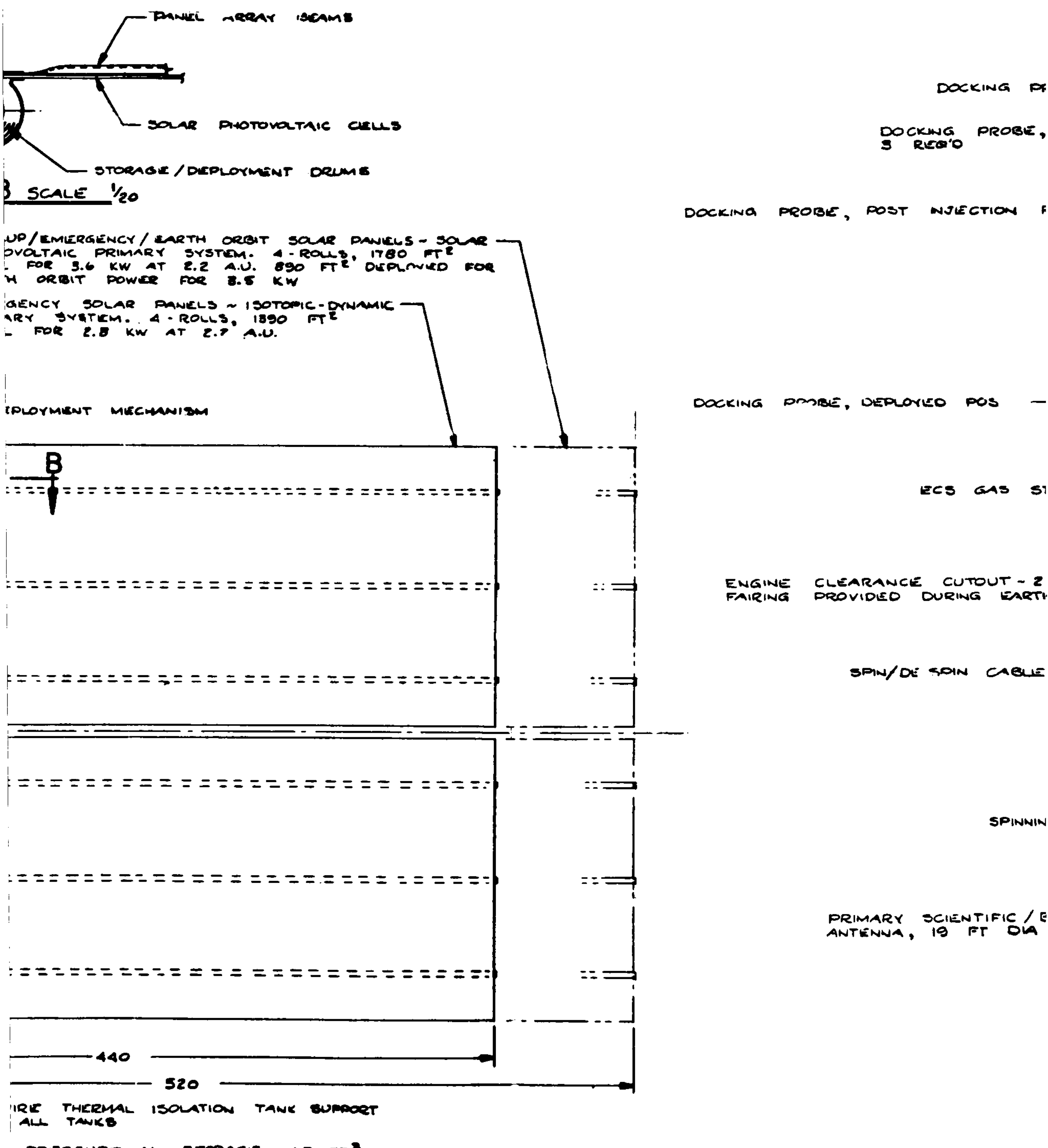

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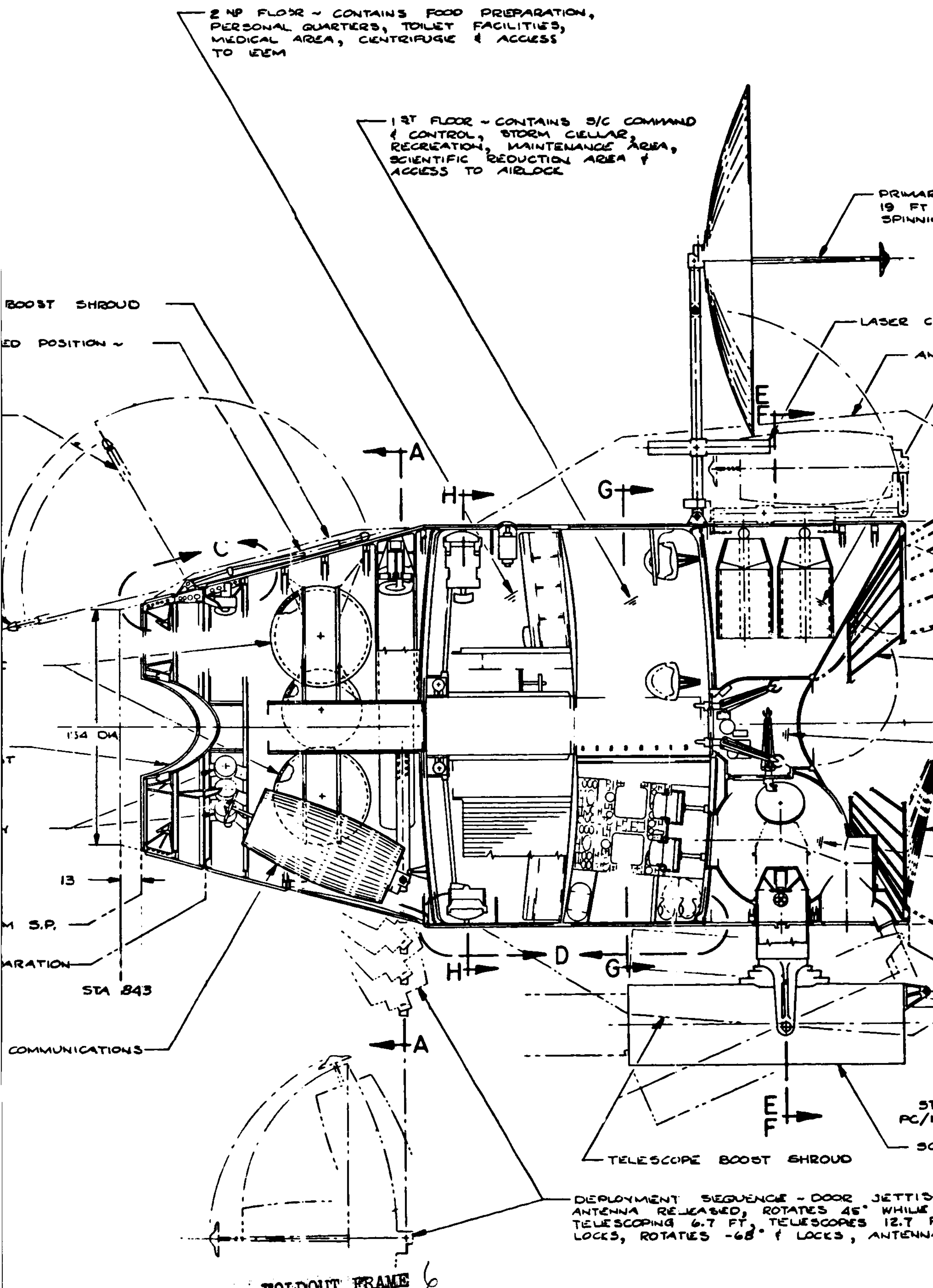

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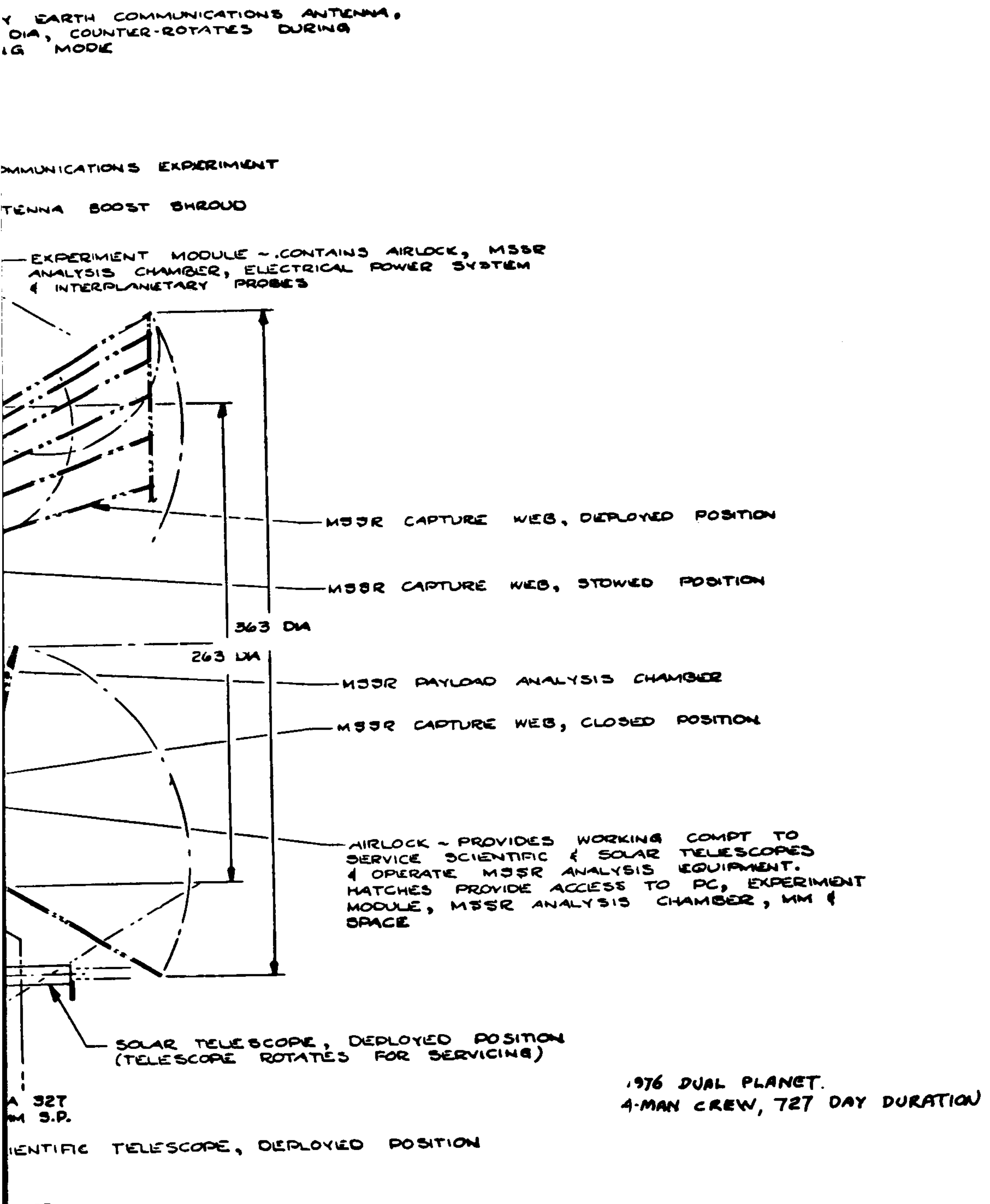

Figure 6.1. Mission Module Configuration

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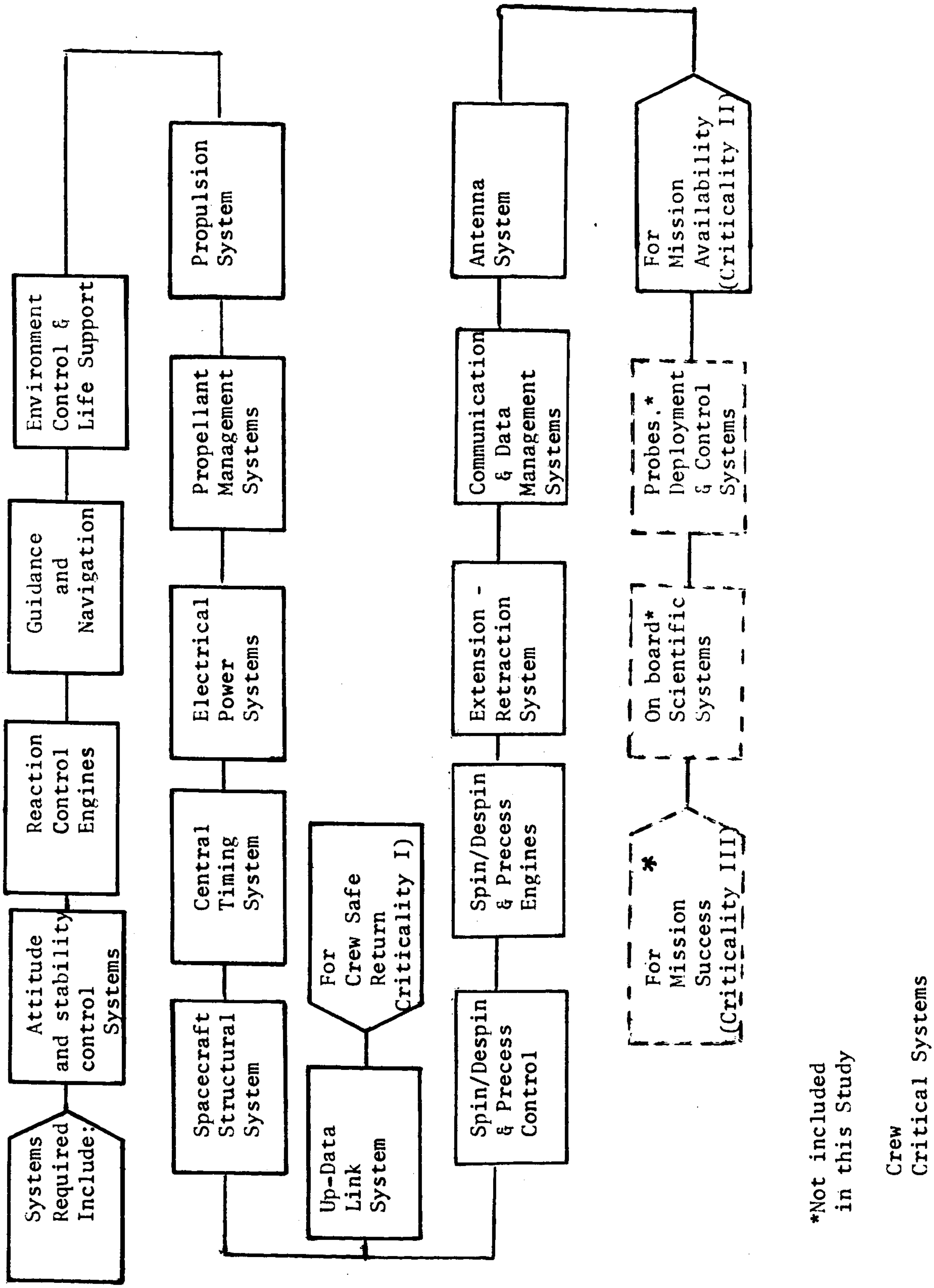

Figure 6.2. Baseline Mission Systems Requirement Logic

Table 6.1. Crew Sensitive Systems Summary (Criticality I)

Section No.	Spacecraft System	Reliability	Spares Required	Contribution to P_s	Spares Weight (lb)	Status*
4.2	Attitude and stability control system	0.879	13	0.997	19.6	Qualified
4.3	Reaction control engine	0.99975	0	0.99975	0	Qualified
4.8	Guidance and navigation	0.7195	44	0.998	137.2	80 percent qualified
4.1	Environmental control and life support	0.104	85	0.9991	165.0	70 percent qualified
4.6	Propulsion system	0.996	0	0.99999**	0	95 percent qualified
4.5	Propellant management	0.989	0	0.999999**	0	85 percent qualified
4.9	Electrical power	0.788	17	0.9995**	103	50 percent qualified
4.10	Central timing	0.978	4	0.999993	8.3	Qualified
4.4	Updata link	0.994	4	0.999	5.0	Qualified
	Structure	0.999999	0	0.999999**	0	Qualified
	Criticality I Totals	0.05	167	0.993	438.1	

*Except for life tests

**Design Redundancy can vary achieved values, see Table 6.3.

Table 6.2. Crew Comfort Sensitive Systems Summary (Criticality II)

Report Section	Spacecraft Systems	Reliability	Spares Required	Contribution to P_s	Spares Weight (lb)	Status*
4.1	ECLSS/O_2 recovery	0.90	18	0.9993	48.6	Unqualified
5.1	Gravity control	0.83	4	0.999	28.2	50 percent qualified
5.2	Communications (MM)	0.38	14	0.99	78.0	90 percent qualified
5.2	Communication (EEM)	0.99	0	0.99	0	Qualified
5.3	Antenna	0.81	41	0.994	3.0	70 percent qualified
5.4	Extension/Retraction	0.9998	0	0.9998	0	Unqualified
4.9	P/O Electrical Power	0.787	14	0.9995	292	Qualified
	Criticality II Totals	0.18	91	0.973	449.8	

*Except for life tests.

Table 6. 3. Required Design Characteristics, Nonrepairable Systems

Report Section	Spacecraft System	Recommended Design Characteristic
4. 6	Propulsion system	1. Use of a three-engine configuration recommended for planetary flights with at least one protected by a blowout cover over the nozzle. 2. Provisions to purge lines and chamber after use.
4. 5	Propellant management (propulsion and RCS)	1. An extra helium supply is recommended which would be isolated from the systems by a gate valve to prevent leakage. A crossover manifold should also be included. 2. Provisions for isolating the pressurant from the feed system are recommended for use during inactive periods. This would include isolation gate valves and bleeder lines.
4. 9	Electrical power system (isotope source)	1. Provide redundant combined rotating unit loops (3 for 7 kw system) since these are impractical to repair. 2. Provide a passive converter, such as the thermal-electric isotope, which would be designed to carry the emergency load.
	(Battery source)	Provide dry-charged, manually activated batteries for the reentry power. (Also applicable to EO logistic vehicle.)
None	Spacecraft structure	Provide an adequate design margin against structural loads and meteroids. Meteroids present the largest hazard but a multilayer design with a weight of about 3. 5 pounds per square foot reduces the risk of one puncture to less than 1×10^{-3}. (Ref 6. 3)

included in the study were the detailed consideration of spacecraft structural elements. These components have been proven to be failure-free when provided with an adequate design margin, that is well within the state of the art. See Space Division studies recorded in References 6. 1 and 6. 2.

6. 2. 2 Earth Orbital Mission Applications

The applicability of the data contained herein to the potential earth-orbital missions would depend on the similarity in functional requirements, particularly those affecting crew recovery. To this end, Space Division performed a study under Air Force sponsorship (Reference 6. 3) wherein an earth-orbital spacestation and an associated logistic spacecraft were defined in detail and analyzed as to emergency characteristics. Many of the systems normally included in the planetary spacecraft are not required for the earth-orbital spacestation, and even the associated logistic vehicle is somewhat less complex. Figure 6. 3 depicts (in logic form) the system functions which affect P_s for both the spacestation and the logistics vehicle. Note that only six subsystems affect crew safety in the spacestation and approximately the same functions for the logistic vehicle operations. However, the logistic vehicle duty cycles are less than 24 hours in duration with shorter dead-storage cycles (duration of 90 days was recommended).

The comparison of such an earth orbital spacestation with the study baseline and determining the resulting effects on the support requirements are reflected in Table 6. 4. The logistic vehicle, in many ways, is analagous to the planetary mission earth entry vehicle and because of the low duty cycles per mission Space Division studies, (Reference 6. 4 and 6. 5) have shown that Apollo level technology will satisfy the requirements.

From Table 6. 3, there are less than four chances in one thousand that the orbiting spacestation will require more than 288 pounds of spares or 115 maintenance or repair actions to assure crew safety. There are less than two chances in one hundred that there will be over 170 maintenance and repair actions or over 660 pounds of spares will be used throughout a two-year spacestation cycle. Both of these estimates are based on the premise that all systems have been qualified for the required life span. No scientific or military systems have been included.

6. 2. 3 System Failure Inferences

The fact that failures will occur on the extended missions seems evident from the data. However, there is no reason to believe the crew will be overloaded with maintenance problems. In fact, the contrary seems more evident. Interpretations of the data can facilitate a better understanding of its meaning.

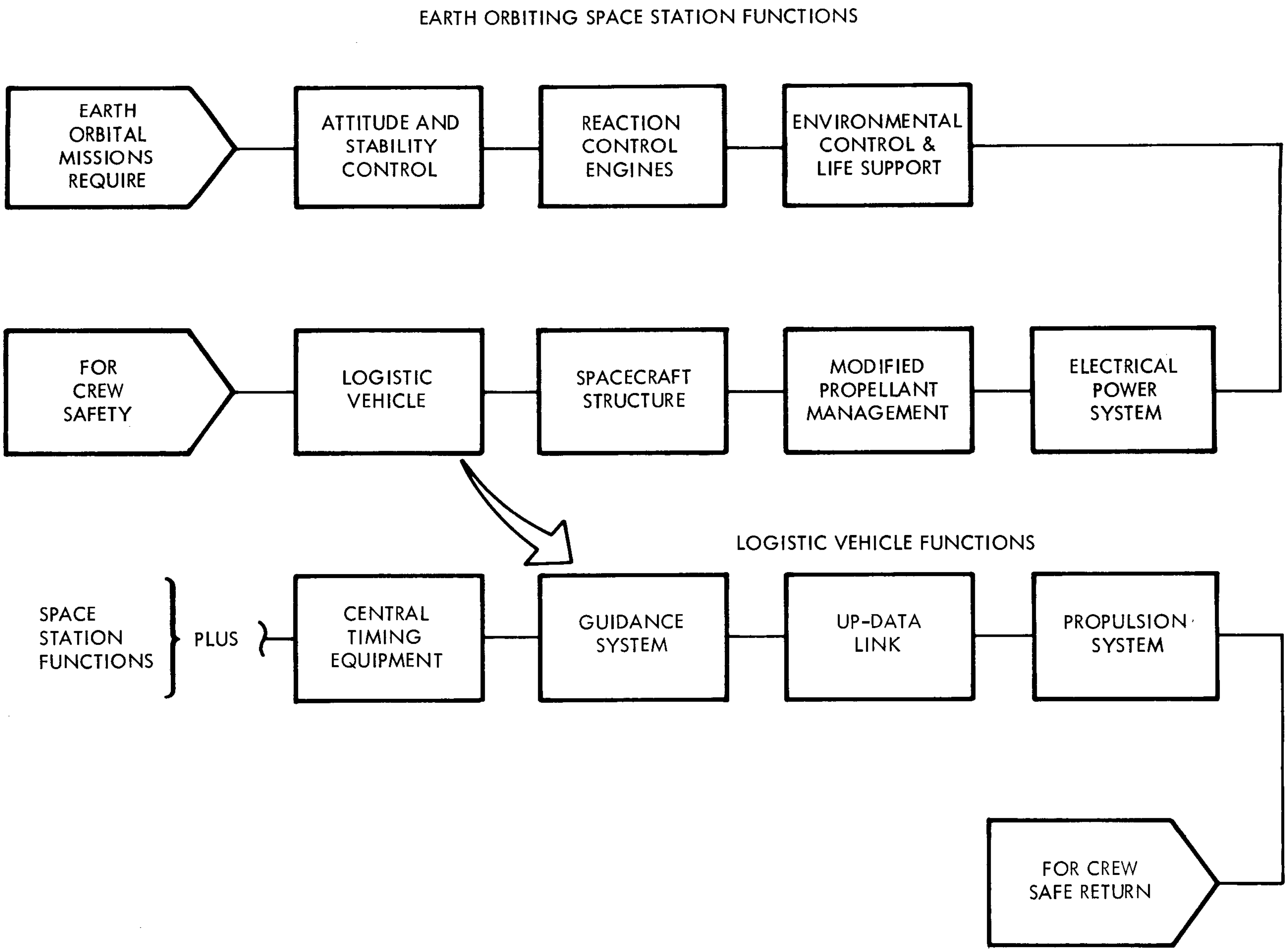

Figure 6.3. Earth Orbit Mission Crew Safety (P_s) Logic

Table 6.4. Earth Orbiting Spacestation, Support Requirements for Crew Safety

Spacestation System	Reliability (2 yr)	Spares Required (2-yr total)	Contribution to P_s	Spares Weight (lb)	Status
1. Attitude and stability control system	0.879	13	0.997	19.6	Qualified
2. Reaction control engines	0.99975	0	0.99975	0	Qualified
3. Environmental control and life support	0.104	85	0.9991	165	80-percent qualified
4. Propellant management	0.989	0	0.999999	0	90-percent qualified
5. Electrical power	0.978	17	0.9995	103	50-percent qualified
Criticality I (P_s)		115		287.6	—
6. Communications	0.38	14	0.99	78	90-percent qualified
7. Antenna Systems	0.99	41	0.999999	3	Qualified
8. P/O Electrical Power	0.787	14	0.9995	292	Qualified
Criticality II		170		660	—

Referring back to Table 6. 1 where the mission reliability was quoted as 0. 05, this means that there is:

95 percent chance of one or more failures, or less than 1-percent chance of more than 167 failures.

Clearly, the crew maintenance and repair work load will be low compared to the other tasks required.

6. 3 DEVELOPMENT PROGRAMS REQUIRED

It is true that the preponderance of system functions are qualified for the planetary missions. However, it is equally true that there are some functions that are not qualified as yet. Again, all the recommended systems have shown feasibility. Tables 6. 1 and 6. 2 also reflect the status of the system function in terms of their demonstrated ability to meet the requirements of the extended space missions. Specific system function study and/or test requirements are listed within the system section of this report. Some of these may be summarized as follows:

1. All system functions require some form of life test, not necessarily to assure life expectancy but rather to identify life limits and characteristics.

2. The life support functions associated with the new functions, such as water recovery, CO_2 removal, and O_2 regeneration, must be developed and qualified.

3. The new source of electrical power, the isotope systems, must be developed and qualified. These systems show great potential both in terms of reliability and as an economic source of power.

4. A new type of antenna system should be considered, such as the phased array, a mechanically rigid structure, in place of a constantly oriented device. (See Appendix 3.) The Apollo antenna will meet the requirements as demonstrated in Section 5. 3; because of potential EVA requirements, it may not be optimum.

5. The systems associated with the extension/retraction for the artificial gravity mode must be better defined, developed, and qualified. The advantages and disadvantages of this type of mode should be studied in detail, from a systems engineering point of view as opposed to the normal human factors approach.

6. The guidance functions are notably weak links in the system's design and, although they will meet the requirements safely, other alternatives may prove to be much more effective (see Appendix 5).

7. High density packaging techniques such as that described in Appendix IV should be developed to simplify future mission maintenance requirements.

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1.2 Manned Mars and/or Venus Flyby Vehicle Study. North American Rockwell, Space Division, SID 65-761 (August 1965).

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4.5 Effects on Crew Safety. North American Rockwell, Space Division, SID 66-872 (May 1966).

4.6 Reaction Control Engines for Long Durations Missions. Marquardt Corp., MIR-109 (August 1967).

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APPENDIX I. OPTICAL USAGE SCHEDULE

During the planet encounter phase, the vehicle attitude must be held to the local vertical to an accuracy equivalent to that required for the Apollo-X mission mapping phase (450 $\widehat{\text{sec}}$ maximum about any axis) when the G&N system is controlling atttiude. Factors affecting the inertial reference attitude are the initial alignment accuracy, the gyro drift, and some vehicle control uncertainty. In maintaining the local vertical with respect to the planet, the trajectory position uncertainty also becomes a factor.

The problem becomes most difficult at periapse, since the local vertical angular error becomes a maximum for a given position uncertainty in the local horizontal plane. For a Mars radius of approximately 1800 n.m. and a periapse altitude of 200 n.m., a one-nautical-mile horizontal position uncertainty would produce a local vertical uncertainty of 100 $\widehat{\text{sec}}$. Combining this with the misalignments produced by one meru gyros, one $\widehat{\text{min}}$ initial misalignment, and some vehicle control system uncertainty, the inertial reference must be updated at a rate of at least once per hour to maintain a one-sigma local vertical of 150 $\widehat{\text{sec}}$.

At the start and ends of planet encounter, the vehicle distance from the center of Mars is approximately 5×10^6 n.m. A position uncertainty of 10 n.m. would yield a negligible local vertical uncertainty of approximately 0.4 $\widehat{\text{sec}}$. Therefore, combining this with the other attitude misalignment uncertainties as above, the inertial reference would only need to be updated after two hours or more.

An average reference alignment update every 1.5 hours was, therefore, assumed for the duration of G&N attitude control during the planet encounter phase. The gyro drift rate and the navigation uncertainty near periapse are optimistic values, but the overall average alignment rate is felt to be more than adequate since it is doubtful that the precise attitude control is necessary continuously and the greatest local vertical accuracy is probably not needed near periapse. It would seem more likely that the greatest vertical holding accuracy would be necessary at the beginning and end of the planet encounter phase when probes are launched and recovered. During these periods, vehicle positional accuracy has very little effect on the ability to hold local vertical plus the fact that ground tracking would be available.

An average navigation schedule of one sighting every two hours has been chosen for the on-board system during the planet encounter phase. This corresponds closely to the optimum schedules determined for the translunar and transearth phases of the lunar landing mission. The total number of sightings (230) also agrees closely with the number determined from the optical sighting schedule given in Table 13-4 of NAA SID-67-549-6-3.

APPENDIX II. MAIN ENGINE PASSIVE PHASE ANALYSIS

From Table 4.36, the coast reliability was calculated to be lower than the desired level. The following potential major causes of failure can be identified:

Meteoroid bombardment

Creep of metals

Materials degradation or sublimation

Propellant exposure

There are two aspects to the potential meteoroid problem. The first is the possible encounter of a micrometeorite cloud, which can cause erosive loss of the emissivity coating on the nozzle extension. The second is the possibility of puncture of one or more parts of the engine by meteoroids. These particles are estimated to have masses ranging from a fraction of an ounce to as high as 10^{24} tons. Of equal concern are the velocities of these particles, estimated to average about 50,000/ftsec and to range as high as 250,000 ft/sec. While some research has been performed in the field of hypervelocity impact, much is yet to be learned about the mechanism of impact, the effects of impact, and the best ways of shielding against impact. However, the nozzle is the most exposed part, and it is estimated that a puncture will have little effect on engine performance. Some additional analysis seems desirable.

There are two primary effects of creep. One is the possible loss of sealing capability at inter-component joints due to the elongation of the bolts, and the other is the possible rupture of one or both of the high-pressure pneumatic tanks or lines as a result of exceeding the creep strength of the metal or because of stress corrosion. An analysis of the existing designs should be made to assure an adequate design margin with respect to creep and stress corrosion. Design changes should be made as deemed necessary. An alternate approach for the case of bolt elongation would be to provide for retorquing of the bolts near the ends of the coast periods. However, as noted in the Table, retorquing of the bolts might not be satisfactory for the metal seal used in the joint between the nozzle extension and combustion chamber; the reason for this is that metal seals generally have to be compressed beyond their yield points in order to achieve effective sealing capability.

The potential problem of materials degradation or sublimation is applicable to three primary areas: the binder used in the combustion chamber, the nonmetallic seals, and the electrical insulation. In general, the long chain polymers found in the nonmetallic component materials have vapor pressures much lower than the 10^{-12} Torr pressure found in space, ultra-violet radiation does not penetrate them deeply, and they are useful at temperatures below 300 F. However, polymers such as nylons and neoprenes show high decomposition rates in vacuum. On the other hand, some commonly used elastomers, such as Teflons, silicones, and Mylar, exhibit excellent behavior in high vacuum. Teflon has excellent lubricant characteristics in high vacuum, but most other polymeric materials are of doubtful value as lubricants in the environment of space.

Optical transmission of polymers will probably be seriously damaged by unfiltered sunlight; the effects on optical absorption and emission are less certain. Electrical and mechanical properties of polymers will probably be unaffected by sunlight, except in the thin outer layer. In general, exposed polymer surfaces will deteriorate rapidly in the earth's radiation belts and perhaps from solar particle emissions. The flexibility, strength, and electrical characteristics of Teflon, nylon, butyl rubber, and similar materials will be detrimentally affected in the radiation belts even through heavy shielding. The actual changes to the materials will depend on the type of exposure, wave length, time of exposure, and type of material. It is therefore extremely difficult to draw conclusions without experimentations, especially in vacuum at elevated exposures.

In some cases, vacuum degrades polymeric systems by volatilizing components essential to the optimum properties of the part. In others, it improves the properties by the same volatilization mechanism wherein impurities are outgassed. The danger of this volatilization process lies in the contamination of electrical contacts and other surfaces that depend upon an uncontaminated surface for essential properties. Since the combustion chamber is the component that would be most affected by this volatilization, it is recommended that a method be developed to seal off the nozzle and maintain a GN_2 pressure within the chamber assembly.

Most of the organic materials used in the SPS engine are long-chain polymeric compounds that degrade in vacuum, not by evaporation or sublimation, but by breakdown of the compounds into smaller, more volatile fragments. Decomposition of organic materials occurs throughout the material and not only on the surface. It is affected by the nature of the particular monomer, by chain length (molecular weight), and by branching and crosslinking of the polymer. Small amounts of impurities can also greatly accelerate decomposition. The molecular weight of these fragments is not well established, and neither is the equilibrium decomposition pressure of most polymers. It is, therefore, necessary to turn to direct experimental studies of the weight loss of polymers in vacuum. It is risky to extrapolate experimental data because, in most studies, exposure time has been limited, and because the curves of weight loss versus time are usually nonlinear. It is recommended that materials degradation information be obtained from examination of similar materials in existing satellites, if possible.

Prolonged exposure to the propellants has the potential for causing a deterioration of the software seals and a formation of salts. While it is true that the hard vacuum of interplanetary space tends to purge the engine of propellants downstream of the bipropellant valve, the propellants will remain trapped within and upstream of the bipropellant valve; it is also possible that propellants adsorbed into seals downstream of the bipropellant valve during an engine firing will remain there after the engine is shut down. Salts can be produced in the bipropellant valve, the lines, or the main propellant tanks, and can cause a clogging of the propellant filters or the injector orifices. Of specific concern is the formation of ferric nitrate, $Fe(NO_3)_3$, as the result of a reaction of nitrogen tetroxide (N_2O_4) and stainless steel. Test experience has indicated that large amounts of contaminants are formed in 20 to 40 days with 24-hour temperature cycles of 50 to 110 F. Further analysis of this phenomenon is required as well as development effort to design propellant containers and lines that will not react with N_2O_4.

A review of Table 2 indicates that the thrust mount is probably acceptable without change. Components of somewhat greater concern are the following:

Combustion chamber

Injector

Gimbal

Probably immediate concern should be directed toward the following components:

Nozzle extension

Propellant lines

Pneumatic actuation system

Bipropellant valve

In the case of the nozzle extension, the main concern is with possible meteoroid impact. Perhaps a shield should be developed to protect the nozzle extension during the long coast periods. However, such a shield would probably have to be retracted from the nozzle extension during periods of engine firing in order to achieve the full effectiveness of heat radiation. The alternative to a protective shield is to carry a spare nozzle for installation subsequent to a failure. The primary concern in the case of the propellant lines is the filtering capability and material storage compatibility with N_2O_4. The design should be carefully evaluated to assure that all contamination can be effectively filtered out of the propellants without impeding the flow of propellants as a result of clogged filters. Further development effort is necessary to resolve the N_2O_4 compatibility problem. The probability of meteoroid damage to the propellant lines, gimbal actuators, and harnesses could be practically eliminated by submerging the engine approximately 12 inches further into the service module and lowering the protective heatshield.

In the case of the pneumatic actuation system, concern must be focused on the tanks, lines, and valves that are continuously exposed to the high-pressure GN_2. The design must be checked and modified as necessary with respect to the potentially adverse effects of creep. The valving should probably be analyzed with the objective of decreasing the chance of external leakage. Perhaps a provision should be made for the recharge of the tanks in the pneumatic actuation system in the event of a slow GN_2 leak. A possible redesign utilizing electric motor actuation system instead of the pneumatic actuation system should also be investigated.

The main areas of concern with the bipropellant valve are propellant leakage and the above mentioned N_2O_4 compatibility problem. Monopropellant leakage would probably result in evaporative freezing within the flow passages of the manifold and could create flow stoppage during subsequent startup. Loss of propellants and possible explosion are other effects of propellant leakage. It is recommended that a design study of possible alternative valve designs be completed with the purpose of reducing the probability of propellant leakage and eliminating long term coast N_2O_4 incompatibility and subsequent contamination formation.

APPENDIX III. ALTERNATE ANTENNA CONSIDERATION

The antenna systems reliability may possibly be further enhanced by utilizing a phased array antenna system. The quasi-redundancy inherent in such systems allows a number of failures before serious degradation in performance occurs. The validity of this assumption would be compromised, of course, unless one takes pains in the original system design to ensure adequate protection against the effect of failure of items that are not naturally redundant, such as control devices, prime power, etc. It is also necessary to exercise care in the design of redundant units for life commensurate with the interplanetary missions within the necessary phase and amplitude tolerances. However, it appears that phase shifting techniques to accomplish electronic inertialess beam positioning offers an intrinsically higher degree of reliability than a servo-driven, parabolic, dish-type antenna. In order to model such a phased array so that a quantitative evaluation of reliability may be conducted for direct comparison with conventional methods, it will be necessary first to define the degree of degradation which could be tolerated in terms of the number of permissible antenna element outages, disregarding contributions to the beam-pattern degradation of unwanted sidelobe level effects due to phase and amplitude errors or tolerances present in the RF signals at the antenna element. In addition, this type of system allows the practice of in-flight (and EVA) maintenance actions, such that the failed items in the array may be located and replaced, thus allowing for the avoidance of additional outages which may occur so as to reach the permissible outage threshold. This may be accomplished by a relatively small number of universal on-board spares.

The model for such a determination entails the determination of system availability given the pertinent parameters. System available for a quasi-redundant system, assuming that sidelobe level degradation is a function of element outage and that propagation of failure from one element to another in the array does not occur, is given by

$$\sum_{i=o}^{N} P(x \geq K, i) = \sum_{i=o}^{N} Pi\ P(x \geq K/i) \tag{1}$$

where $P(x \geq K)$ is the probability that sidelobe level is greater than or equal to K, where K is some minimum acceptable sidelobe level (x), and N is the total number of array elements; thus, the availability is found by summing

P(x Ki) the joint probability that exactly x elements are out and sidelobe level is greater than, or equal to, K over all possible values of i, and, similarly,

$$\sum_{i=o}^{N} P(x < K, i) = \sum_{i=o}^{N} Pi\ P(x < K/i) \qquad (2)$$

where $P(x < K)$ is the probability that sidelobe level is less than K. Simple approximations to Equation 4 and 5 are given by

$$P(x \geq K) \sim P(x \geq K/M) \sum_{i=o}^{M} Pi \qquad (3)$$

and

$$P(x < K) \sim P(x < K/M) \sum_{i=o}^{M} Pi \qquad (4)$$

when $\sum_{i=o}^{M}$ Pi is near unity and $P(x \geq K/M) > 0.99$ or, conversely, $P(x < K/M) < 0.01$. These are conditions which may readily be placed on an array with stringent requirements that take the following form; 99 percent of the time the array must have a sidelobe level at least as great as K. Equation 3 is simply the product of the probability that the sidelobe level is greater than K, given exactly M elements out, times the probability that M or less elements are inoperable, and this result is a good approximation to the probability that the sidelobe level is greater than K.

In order to compute availability given by Equation 3, it is necessary to compute each of the two terms involved, one a conditional probability and the other a summation of unconditional probabilities. The conditional probability, $P(x \geq K/M)$, may be called out by the antenna system engineers who determine some threshold value M, taking into consideration the statistical distribution of degradation with a given outage level. The maximum number of simultaneous outages that can be tolerated for a given fraction of the time is represented by M; both x and M are sensitive to the geometry of element outages in the array face and make the statistical description of the threshold necessary. Naturally, M cannot be increased to an appreciable fraction of the total number of elements without resulting in severely degraded performance.

In a practical high-performance array, M is usually kept sufficiently small, relative to the total number of elements, so that beam directivity error and the deficiency of radiated or received power are not as important contributions to unacceptable performance as the sidelobe level degradation due to the M outages. Considering an array face of N elements, the probability that exactly i elements are out may be shown to be

$$P_i = \frac{\frac{N!}{(N-i)!} a^i}{\sum_{i=o}^{N} \frac{N!}{(N-j)!} a^j}, \quad N > 0, \ a > 0 \tag{5}$$

where λ is the failure rate per element, u the repair rate per element, and $a = \lambda / u$ (load factor).

When $N > 100$ and $Na < 1$, Equation 5 simplifies to,

$$Pi \sim (1-Na)\,(Na)^i, \quad N > 100, \ Na < 1 \tag{6}$$

This form for P_i may be used in Equation 6; however, various maintenance procedures and system configurations would dictate the exact model. Subsequent studies should be directed toward establishing the quantitative values for failure and repair rates based upon preliminary data on EVA and, together with more refined system design concepts, a rigorous comparison may be made between the array and dish reliabilities for manned interplanetary missions.

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APPENDIX IV. PACKAGING CONCEPTS FOR ELECTRONICS ON LONG-DURATION MISSIONS

This section discusses results of an SD study for effecting required improvements in mission reliability of electronic systems through consideration of packaging. Three levels of circuit integration are considered and compared: (1) discrete circuits, using standard transistors, resistors, etc., (2) integrated circuits, and (3) large-scale integration (LSI) circuits.

Previous studies conducted for Apollo showed a lack of universal modules at the subassembly or replacement level; thus, great complexity resulted with regard to failure isolation and numbers and types of spares to be carried. This complexity eventually led to discarding the in-flight maintenance concept, since added built-in redundancies could be effected at a lesser weight penalty than that incurred by the weight and complexity of an in-flight test system needed in conjunction with in-flight maintenance of heterogeneously designed modules. Hence, the study was aimed toward defining a universal modular concept, based on the state of the art expected in the electronics industry in the 1975 to 1977 period. As shown, the highest degree of universality can be accomplished through the selection of LSI. This will permit multiple applications of module commonality and interchangeability. This, in turn, will allow a 1000-to-1 reduction in total parts count from the discrete circuit concept, allowing much more volume for sparing electromechanical assemblies (e.g., gyros, servo torquers, synchros, resolvers, etc.) and electronics subsystems, as well as such items in which breakthroughs cannot be anticipated (e.g., traveling wave tubes).

While the recommendations contained herein would not allow maximum application of Apollo-type electronic equipment, maximum increases, in concert with required levels of reliability, can only be realized with the recommended changes.

SD studies show that use of Apollo- and AAP-type electronics have a predicted mission completion probability for the Mars flyby mission of less than 1 percent, increased to 30 percent with selection of additional, feasible redundancies. These results point up the need for an in-flight maintenance concept for enhancing mission completion, as well as a packaging concept to permit in-flight maintenance. Large-scale integration (LSI) circuits appear to be the most favorable solution. This section delineates such factors of LSI design as volume, failure rates, built-in passive redundancy, multiusage packaging, interchangeability, and circuit intraconnections to point up the feasibility of selecting the LSI concept to realize the mission reliability requirements for future Mars missions.

LSI Concepts

The evolution of highly complex monolithic integrated circuits has led to refinements in component size, tolerances, and functions which make feasible the integration of entire subsystems within one package. This development, which has been termed "large scale integration" or LSI, presents a step from the integrated circuit to the integrated subsystem. Furthermore, as circuits become more complex and integrated subsystems become more widespread, a trend to circuit standardization and multiusage packages (i.e., interchangeability and universal spares) is inevitable. This has significant meaning to realistic packaging concepts for manned, long-duration missions, where previous studies have shown a need for in-flight maintenance as a means of assuring mission completion.

In the semiconductor industry, the trend is toward structures having more complex arrays in which hundreds of circuit functions are interconnected on a single chip. This approach provides lower power and higher reliability. The power is reduced primarily because interconnection capacitance is reduced. Reliability is primarily increased because there is an order of magnitude reduction in components and connections. The array-type structures of LSI are being produced in limited quantities, and specific reliability data have been collected from Fairchild and Texas Instruments and IBM, Refs. 4-3, 4.4.

It is expected that by the 1975-77 time period, LSI will be a common and proven technology. Presently there are two device approaches available which are compatible with LSI technology: metal-oxide semiconductors (MOS) and minority charge carrier (bipolar) integrated circuits. The controversy over which technique, MOS or bipolar, is superior for LSI has continued for several years. MOS advocates claim increased circuit density, lower power consumption, higher power supply efficiency, simplified manufacturing procedures, and higher reliability. Some of these claims are controversial. For example, MOS transistors require fewer process steps, but some of these steps may be more critical than their bipolar counterparts. The bipolar transistor shows a clear superiority on a speed-power basis. This results from the bipolar transistor's superior transconductance per unit area, which in turn is a measure of the device's capability to charge and discharge capacity. In addition, bipolar arrays will be compatible with today's existing integrated circuit signal levels, power, and logical organization. However, for digital circuits, the MOS techniques appear to be the more promising, especially from the standpoint of reducing circuit complexity. MOS arrays can be produced as monolithic systems or on an insulating substrate (heteroepitaxial technology).

For space applications, MOS should prove to be the technology of the future for a number of reasons. Its power drain is significantly lower than that associated with bipolar circuits; the higher yields and much smaller size of MOS enable MOS chip complexities to be increased over those of bipolar chips. It is anticipated that the increase in density offers the potential of a greater reliability, since fewer packages and intraconnections are necessary in a given system; data tend to point to the fact that the number of packages in a system and not the complexity of the packages have the dominant effect on system reliability — as is shown in following paragraphs. The only present disadvantages associated with MOS circuitry compared to bipolar circuitry are its slower speed and lower radiation resistance. The speed difference should be somewhat overcome by the development of complementary circuitry; very high radiation resistance is not anticipated as a requirement for the manned interplanetary missions. Figure A-1 shows the anticipated level of LSI complexity (Reference 4-5) over the next ten years for MOS and bipolar arrays. As shown, the current penalty (in terms of package density) for selecting bipolar arrays should be eliminated by the mid-1970's.

LSI Reliability Considerations

The reliability of LSI subsystems has been found (Reference 4-6) to be a function of the number of circuit chips. For example, the overall mean-time-between-failure (MTBF) of such a subsystem would decrease proportionately to the increase in the number of chips as predicted in Figure A-2, Curve A, assuming an MTBF of 10^8 hours for a single chip. This curve represents a theoretical upper limit of MTBF, assuming no additional failure mechanisms are operative as the number of chips increase; Reference 4-3 indicated that some new failure modes have been observed with LSI, but details are considered proprietary in nature and could not be obtained. The actual curve may fall off rapidly with an increase in the number of chips because of the additive influence of interconnection reliability. Curve B which predicts MTBF of multichip subsystem bonds only is derived from Curve A, assuming that 25 percent of the failures of single chips can be ascribed to one (or more) bond failures. Curve C predicts the MTBF of a circuit chip as a function of the number of bonds per chip, assuming that one bond failure amounts to a chip failure. Curve C is derived from Curves A and B which are related to individual chips containing 14 bonds. The predicted subsystem MTBF is uniformly displaced downward as the number of bonds per chip increases, as illustrated by Curves A' and B' corresponding to 100 bonds per chip. Curve C has been observed for thermal compression wire bonds; that is, the increased circuit chip complexity requiring n number of bonding sites only has second order effects on the overall chip reliability.

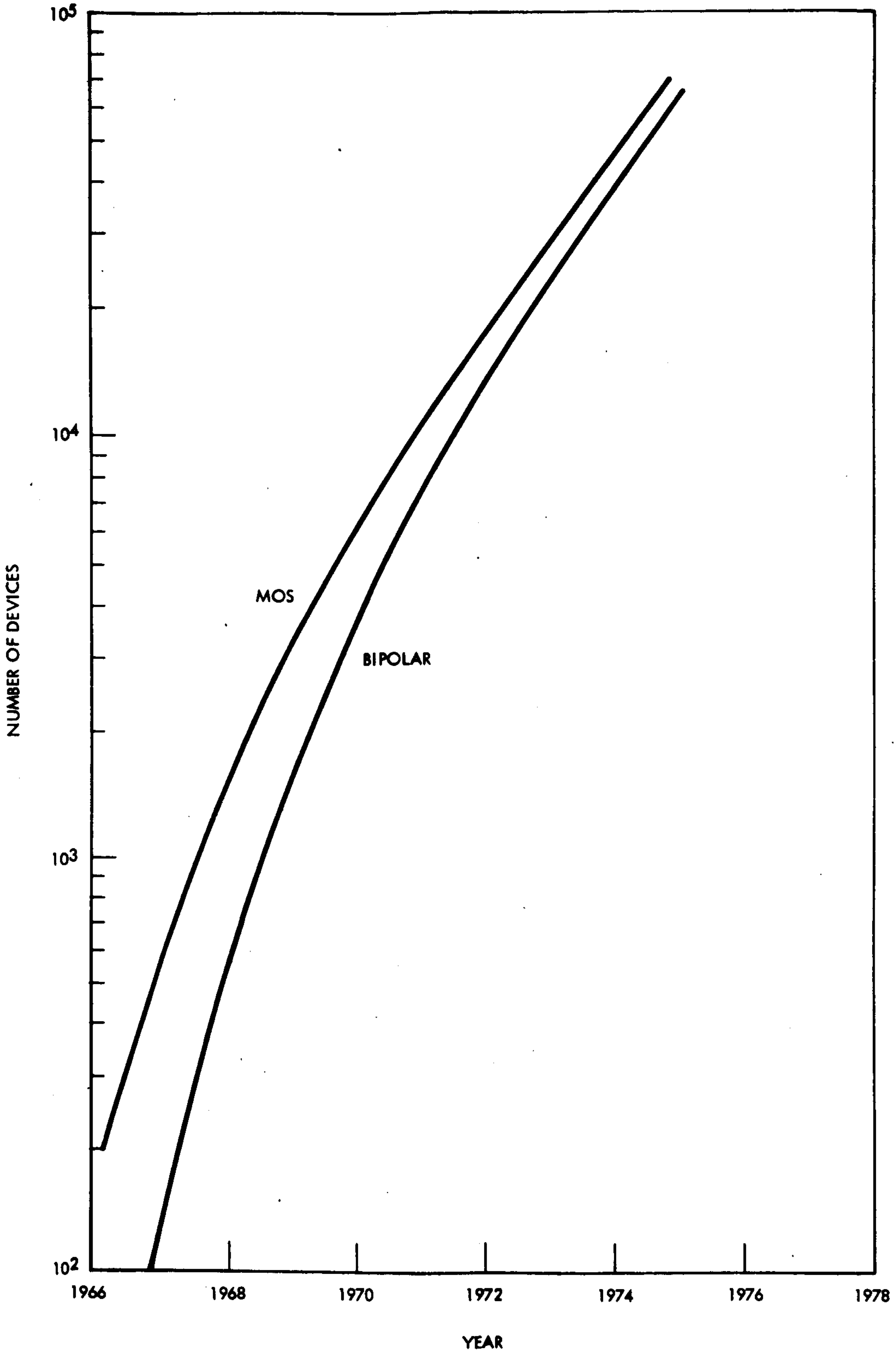

Figure A-1. LSI Density Forecast

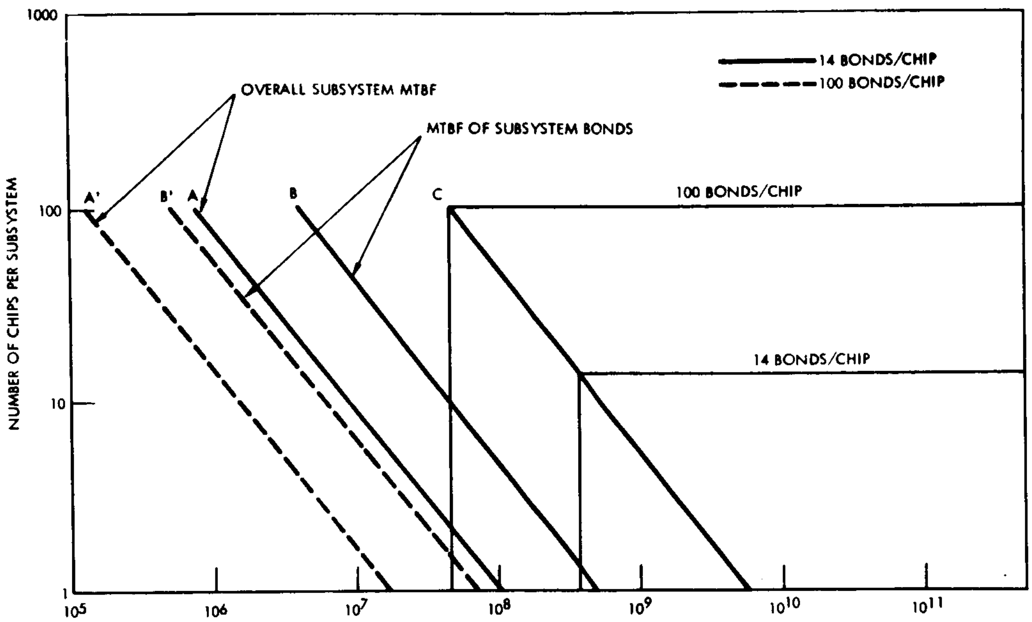

Figure A-2. Effect of Number of Interconnections on Subsystem MTBF

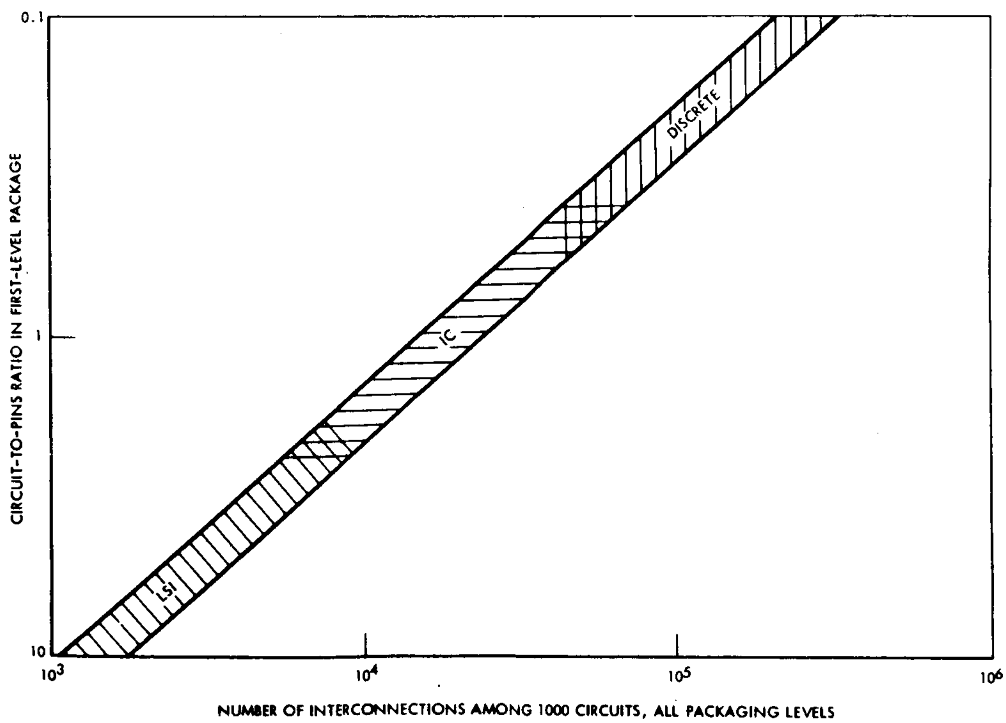

Figure A-3. Effect of Integration Scale on Packaging Scheme

Thus, the simple inverse relationship shown in Curve C is realized. In practice, no difference in reliability has been detected for chips ranging from 10 components each to 160 components each.

A "flip-chip" bonding technique, which effects the simultaneous formation of pure metallurgical bonds between external leads and internal wire bonding sites of a circuit chip, can be used to replace wire bonding. This makes it possible to package several interconnected flip-chip bonded complex-circuit chips in one package. This technique will result in curve C of Figure A-2 being shifted upward on the MTBF axis and the MTBF of the subsystem bonds becoming less dependent on the number of bonds per chip. The shifting of curve C thus shows that the cumulative bond strength of the flip-chip surpasses the single-bond strength of the wire bond. The overall predicted effect of using flip-chip bonding is improvement of the MTBF of the entire system. Further improvement can be obtained by designing redundancy in at the integrated circuit level, as discussed in the following paragraphs.

It is expected that an average of 2 mils center-to-center spacing will be used for 1975-77 MOS circuits. This gives approximately 5,500 field-effect transistors (FET's) per 150 mils square. A few processing breakthroughs enabling yields to be increased would make larger chips available. For example, a 200-mil square chip (or larger) may well be usable in the 1975-77 period. Such a chip would contain 10,000 FET's. These estimates are not critical as to whether the chip is 150 mils square or 200 mils square; the only fact of importance is that a small chip with 5,500 devices can be produced with reasonable yields. MOS chips are presently packaged in forty-lead packs. This situation should be improved with future packaging methods. As an aid, for example, lines could be fanned out on the substrate so that larger packages could be used. Again, it is difficult to predict the development of the packaging technology, but it should be reasonable to expect 100- to 150-pin packs for MOS chips in the 1975-77 period.

Figure A-3 shows the relationship between the number of interconnections and the circuits-to-pins ratio at the first level of assembly for various integration scales from zero integration, i.e., discrete components, through conventional IC's to LSI. Naturally, as the circuit density at the first level increases, the number of interconnections at all levels of packaging decreases, resulting in increased reliability.

Due to some of the inherent limitations of MOS circuits discussed above, it can be assumed that the future will see circuits which combine the best features of MOS and of bipolar structures so that the optimum component number for combined (but monolithic) circuits will be between those given for homogeneous MOS and bipolar circuits. As Figure A-1 indicates, MOS circuits will have a lead over bipolar circuits as far as component number per circuit is concerned, primarily because MOS structures do not require individual isolation.

As an example, a redundant NAND gate is shown in Figure A-4. This may be diffused into one monolithic chip of silicon (Reference 4-7). This type of operational redundancy is best implemented passively. Active failure detection raises the level of complexity — by means of onboard displays as well as a larger telemetry subsystem for monitoring system status — and does not necessarily result in more reliable systems. Passive redundancy is mechanized by performing the same function in parallel paths; a major problem lies in properly combining the outputs of the parallel paths.

Multi-Usage Packing: Interchangeability

The most obvious application of LSI to the manned interplanetary vehicles is in the on-board computer(s), since digital circuitry is generally more amenable to LSI than linear circuits. There are three general configurations for partitioning LSI for digital applications: unit logic, lead-reduction logic, and functional logic. Ignoring the possible impact of wiring rules, the mode used in packaging a particular machine need not alter the logic implementation of the machine. The configurations differ in how interconnections are distributed between the several levels of packaging. In most cases, these configurations are not observed in their pure form. In unit logic, all inputs and outputs for each logic gate are brought to the next higher package level. Early integrated circuits and the discrete component cards used in computers of the early 1960's were packaged in this manner. In lead-reduction logic, inputs and outputs of several circuits are combined before penetrating the next package level. The more dense IC's which will be available in the 1975-77 time frame can be packaged in this manner. In functional logic, entire logic nets are completed prior to penetrating the next package boundary. A few IC's, such as shift registers and adders, are available today packaged in this configuration. Table A-1 presents a preliminary listing of candidate equipment items, by subsystem, for circuit changes using LSI's.

Figure A-5 shows the effect of increasing multiusage on the total numbers of packages in a computer subsystem (Reference 4-4). As can be seen, the system design challenge is to define larger functional blocks which will have multiple usage for greatest in-flight maintainability. Thus, in the 1975-77 time period, Space Division systems engineering personnel (in lieu of component engineers) must interface with a different world than the world of component manufacturers.

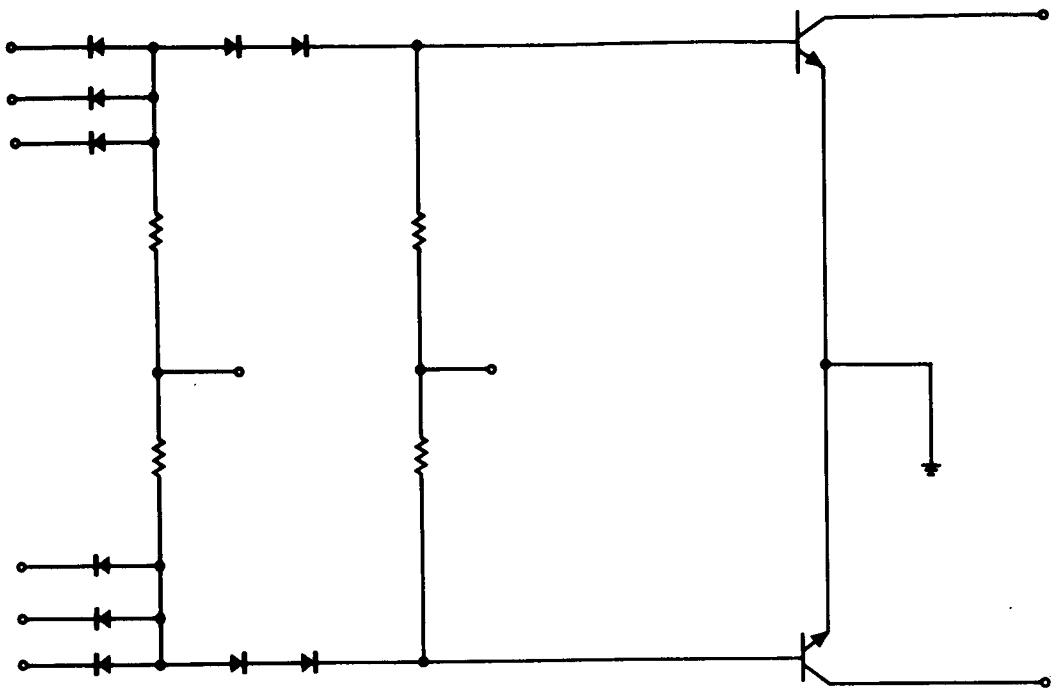

Figure A-4. Typical Passive Redundant Circuit

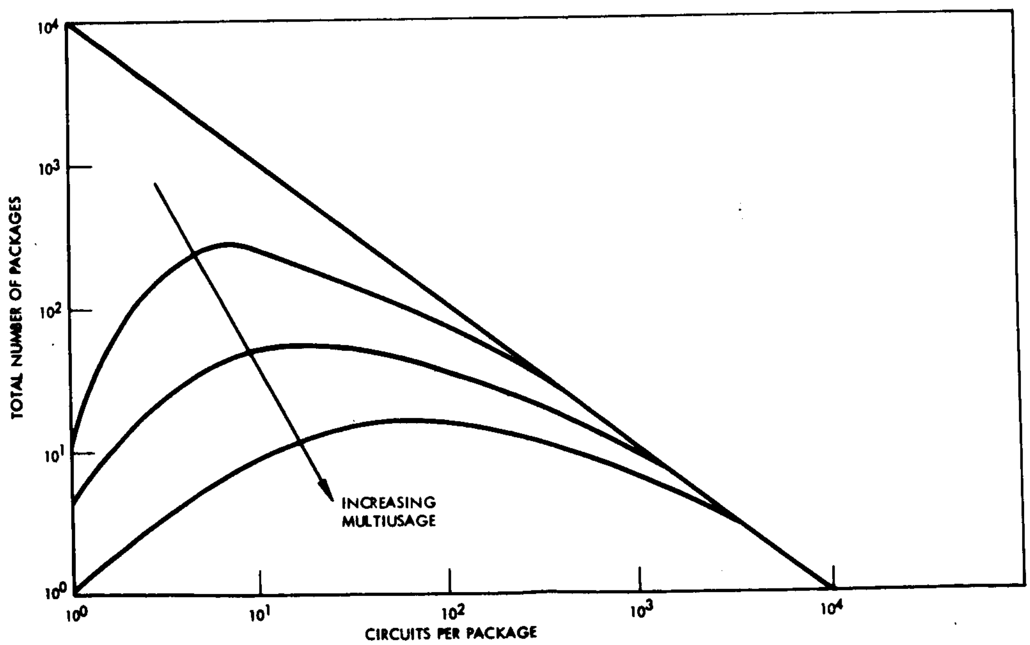

Figure A-5. Effect of the Universal Module in Reducing Circuit Complexity

Table A-1. LSI Candidates by Subsystem

Subsystem	Candidate Equipment
1. Stabilization and Control	BMAG electronics Gyro display coupler GPI electronics Electronic control assemblies (three-axis) Thrust vector servo amplifier Entry monitor system
2. Guidance and Navigation	Power and servo assembly PIPA electronic assembly Electronic coupling display units Command guidance computer Display keyboard Signal conditioning assembly
3. Communications and Data	Pulse code modulation telemetry
4. Instrumentation	Signal conditioning equipment Central timing equipment

LSI Packaging Schemes

Connections to external leads and the final package scheme can be accomplished by extension of current packaging techniques. In one method, a Kovar lead frame is solder bonded to the copper-aluminum interconnection. The assembly is then completely encapsulated in plastic. An alternative scheme uses a ceramic base with glassed-in leads which are lapped flush with the ceramic surface. The terminal ends of the interconnection network overlap the lead ends. A glassed lid is fired on to complete the package. Material selection for packages is governed by the need of the package to dissipate the larger amounts of thermal energy associated with the more complex subsystem packages. Principally because of their higher thermal conductivity, ceramics and metals are usually favored over glass and plastics.

Various module interconnection techniques have been developed (Reference 4-8). Figure A-6 shows the maintainable electronics component assemblies (MECA) system produced by AMP, Inc., which is a versatile packaging system in that it can accommodate most of the existing basic functional circuits and is adaptable to future developments in submodule packaging. The AMP-MECA system relies on its basic building block or cell

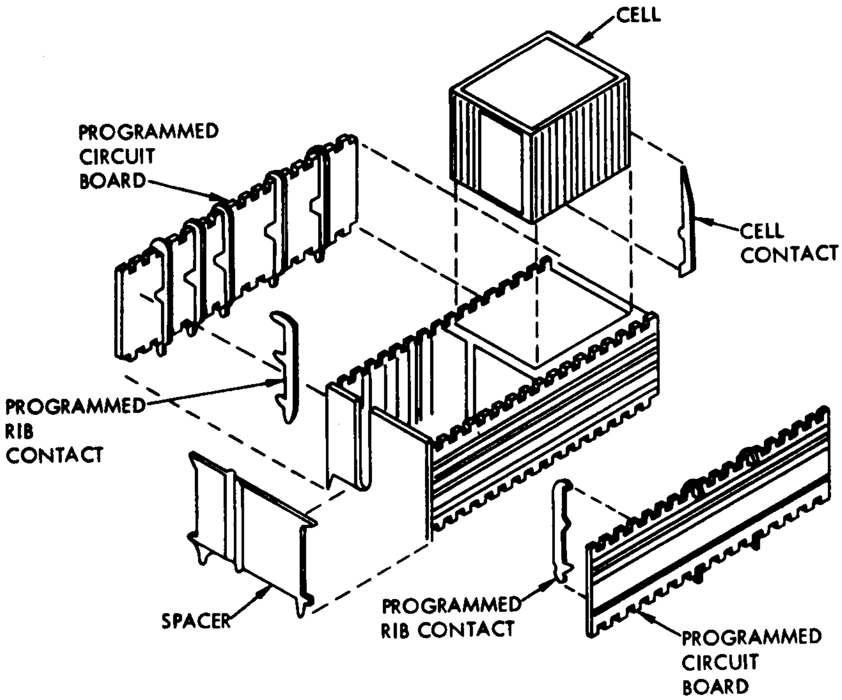

Figure A-6. AMP-MECA Interconnection System

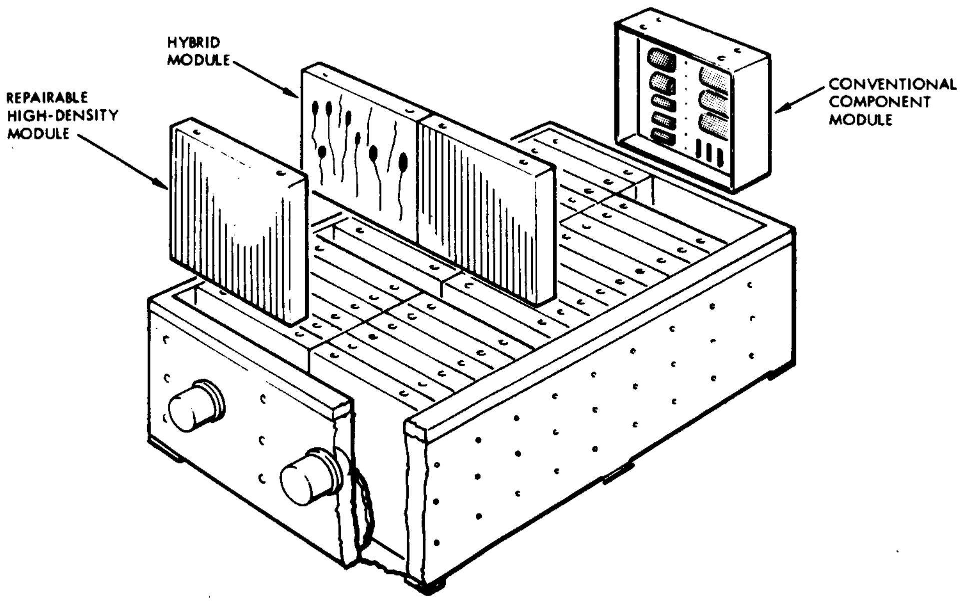

Figure A-7. Boeing Module Housing With Modules Installed which consists of a diallylphthalate box with no top. The circuit module is placed in this cell with contact areas at the top. Contact to the circuitry is made by cell contact clips that run the entire height of the cell. Connection between cells is made by copper strips which run the length of phenolic boards. The strips run perpendicularly to the cell contacts. Rib contacts slide into the cell contacts and make contact with the copper strips through holes in the phenolic boards. These rib contacts may make connections with any combination of copper strips on the board. Preprogrammed interconnection between cells may be made in this manner. Boeing has developed a method (Reference 4-9) of modularizing flat-packs using spring-tempered, thin gauge, phosphor bronze etched sheets; these sheets are assembled into a rigid structure using tabs and slots – a method developed in the toy industry. To install the flat packs into the modular system, it is necessary only to insert the flat-pack into its designated compartment. The flat-pack leads are aligned or registered with the corresponding flat-wire finger protruding from the flat-wiring assembly. The connection methods can be either welding or pulse soldering. The electronic package which houses the modules is constructed using the tab-and-slot concept employed in the module fabrication. The module housing consists of 0.016-inch phosphor bronze sheets which have been etched and solder coated to allow assembly by bending the tabs and fuzing the solder. Double walls are used to rigidize the structure. The electronic package is designed for mounting on a cold plate; a solid copper rim is utilized for the dual function of heat transfer and mechanical attachment to the cold plate. Figure A-9 shows a typical module housing with several modules installed. Other packaging schemes, such as Westinghouse Micpak (Reference 4-10), Surface Advanced Development Module (Reference 4-8), the Amelco Mema (Reference 4-11), etc., should be investigated in order to determine that scheme most suitable for the manned interplanetary vehicles.

Mission Reliability Considerations

A preliminary estimate of the effect of LSI on vehicle system reliability may be made by considering, first, the contribution to the total system failure rate of each class of discrete components for the Apollo electronics system (The Apollo subsystems are composed of discrete components, except for the control timing equipment), and then considering the effect of the longer-duration missions (approximately 700 days), and the effect of increased system complexity (approximately 50 percent) on this estimate. These considerations are diagrammed in the histogram (Figure A-8).

Figure A-9 shows the Apollo parts count for the electronics subsystems together with the projected parts counts for three concepts for the Mars vehicle electronics. As can be seen, the use of LSI will afford a 1000-to-1 reduction in total parts count from the discrete components version, allowing

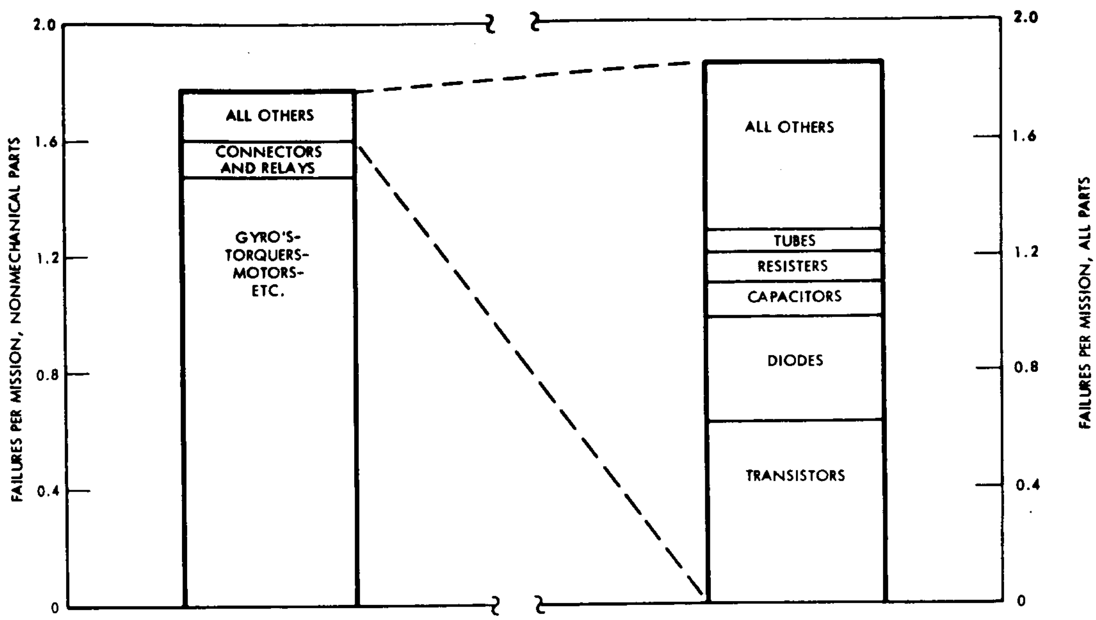

Figure A-8. Apollo Part-Failure Distributions

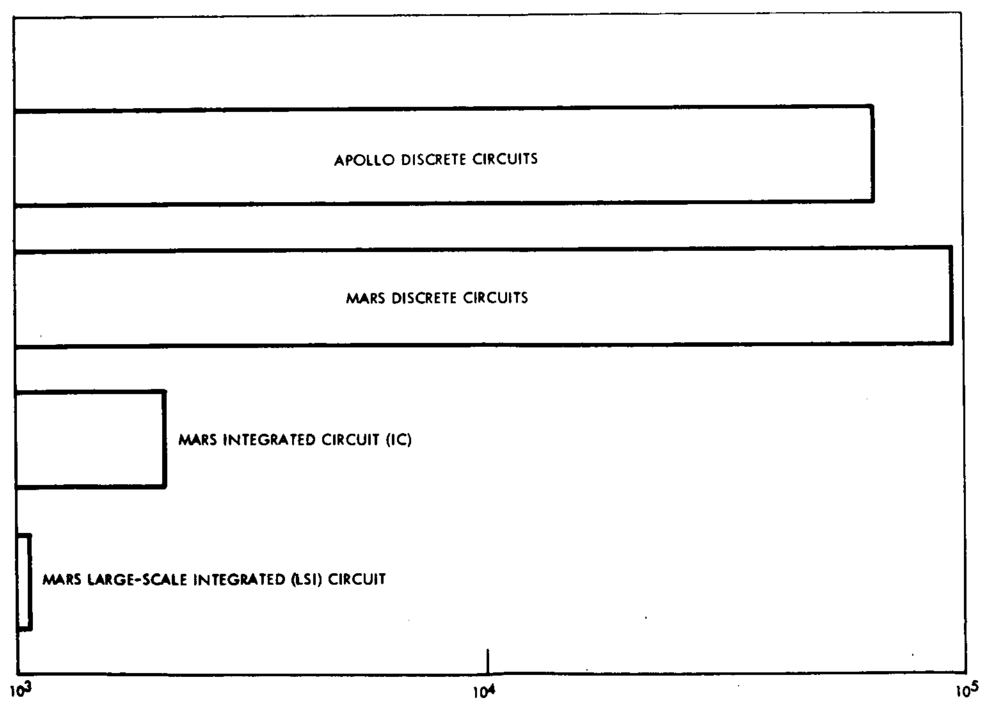

Figure A-9. Effect of LSI on Mars Vehicle Electronics Parts Count much more volume for sparing of electromechanical assemblies (gyros, motors, relays, traveling wave tubes).

Conclusions

Large-scale integration of the electronic subsystems provides significant benefits to vehicle design for long-duration missions. Significant reductions can be realized in weight, volume, numbers of inter- and intra-connections, numbers of unique items, etc., with a corresponding allowance for additional sparing for electromechanical and thermionic assemblies which are inherently unreliable, and for which no significant advance in technology is anticipated in the required time period. Additionally, the reduction in numbers of unique items will allow a maximum degree of interchangeability for in-flight replacement of the electronic modules.

While the recommendations contained herein would not allow maximum application of Apollo-type electronic equipment, maximum increases, in concert with required levels of reliability, can only be realized with the designated changes.

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APPENDIX V. ALTERNATE G&N INERTIAL PLATFORM

One of the critical items of guidance equipment is the inertial platform. A possible means of uprating the Apollo IMU is the gas-bearing gyro used in the Autonetics N-17 system. This type of gyro has essentially no wearout characteristics and a high reliability of about 100, 000 hours MTBF, compared with the Apollo IMU of 772 hours. The majority of failures associated with gas-bearing gyros are related to start-up, and may be virtually eliminated by continuously running them throughout the mission if the power is not excessive; power may be conserved somewhat by running at slower speeds during coast periods, as the operation under acceleration is impaired by slow running speeds. The original estimate for the duty cycle of the IMU was 825 hours; thus, with continuous running in order to preclude the wearout problems with start-up, the reliability of each gyro for the entire 16, 801 hour mission would be:

$$R = {}^{-(10^{-5})\,(1.6801 \times 10^{4})} = 0.821 \qquad (1)$$

compared with

$$R = {}^{-(0.001293)(825)} = 0.344 \qquad (2)$$

for the IMU. The overall enhancement of mission reliability would be given by the differential between two parallel (i.e., one in standby) IMU's compared with some number of gas-bearing gyros of equivalent weight; i.e., nw, where w is the weight of the gas-bearing gyro, and nw is the weight of the IMU,

$$R_{diff} = \left[1 - \left((1 - \ell^{-\lambda_1 t_1}\right)^n\right] - \left[{}^{-\lambda_2 t_2} + \lambda_2 t_2 \, \ell^{-\lambda_2 t_2}\right] = 26\% \qquad (3)$$

where λ_1 and λ_2 are the failure rates of the gas-bearing gyros and IMU, respectively, and t_1 and t_2 are the respective duty factors, assuming operational redundancy for the gas-bearing gyros and standby redundancy for the IMU, and the weight of the IMU is 42 pounds, and the gas-bearing gyros are assumed to be 26 pounds (and n, therefore, becomes 1.6).

An inertial platform (using gas-bearing gyros) similar to that used in the Minuteman system may be configured for the interplanetary vehicle. The platforms may be mounted side by side with the outer (roll) gimbal axis aligned with the command module's reentry roll axis (i.e., 33 deg away from the command module's axis of symmetry). The inner gimbal axis is the pitch axis, followed by the redundant roll gimbal axis and the yaw axis. The platforms may be essentially the same as the basic Minuteman platform except that the gimbals will have to be enlarged to accommodate two-speed synchro readouts on the gimbal axis. Each stable element would include three velocity meters and two gyros. Although envelope restrictions do not allow the use of an alignment block, four separate two-axis bubble levels may be mounted on the stable element for use in ground checkout and calibration operations. Errors of the order of 10-second rms may be produced by this alignment-set arrangement. The weight estimate of 26 pounds each in the expressions derived to compare the reliability enhancement (s.v.) is based upon the use of aluminum gimbals and platform structure. However, it may be necessary or desirable to use beryllium in order to meet the system performance objectives during the launch and reentry portions of the mission. (It should be noted that the outer gimbal axis is 33 deg away from the major thrust axis during the launch and coincident with the axis of drag during reentry.) The outer roll, pitch, and yaw gimbal axes would have on-axis mounted two-speed (electrically) synchro readouts.

It is estimated that the platform angle readouts will provide attitude information with an accuracy of 20- to 25-second rms (this includes readout instrument errors, gimbal axes non-orthogonality, and initial adjustment errors). In addition to the resolvers and pick-offs for normal four axis servo control, a resolver (on the outer roll axis) is required to put the platform pitch and yaw error signals in vehicle stability axes coordinates. Also, a fixed angle resolution of the resulting roll and yaw errors is provided within the system electronics to convert the error signals from stability to launch vehicle coordinates during all phases except Earth reentry. It is anticipated that the modified platform would be 9-7/8 inches by 10-11/16 inches in diameter. Extensions of 9/16 inches and 4-1/4 inches on the ends of the egg-shape (i.e., along the roll axis) would house a d-c torquer motor, the two-speed synchro readout, and the one-speed resolver for the attitude error signals.